LEVEL II

12

14 USAAEFA ~~PROJECT NO.~~ 80-03

AD A098922

ENGINEERING DESIGN TEST 4
YAH-64 ADVANCED ATTACK HELICOPTER

9 FINAL REPORT

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REPORT DOCUMENTATION PAGE

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1. REPORT NUMBER: USAAEFA NO. 80-03 ✓
2. GOVT ACCESSION NO.: AD-A098 922
3. RECIPIENT'S CATALOG NUMBER:
4. TITLE (and Subtitle): ENGINEERING DESIGN TEST 4 YAH-64 ADVANCED ATTACK HELICOPTER
5. TYPE OF REPORT & PERIOD COVERED: FINAL REPORT-Nov 10-29, 1980
6. PERFORMING ORG. REPORT NUMBER:
7. AUTHOR(s): GARY L. BENDER, VERNON L. DIEKMANN, LARRY B. HIGGINS, JOHN D. OTTOMEYER, BARTHOLOMEW D. PICASSO, III
8. CONTRACT OR GRANT NUMBER(s):
9. PERFORMING ORGANIZATION NAME AND ADDRESS: US ARMY AVIATION ENGINEERING FLIGHT ACTIVITY EDWARDS AFB, CA 93523
10. PROGRAM ELEMENT, PROJECT, TASK AREA & WORK UNIT NUMBERS: 46-1-P2052-01-46-EC
11. CONTROLLING OFFICE NAME AND ADDRESS: US ARMY AVIATION RESEARCH AND DEVELOPMENT COMMAND 4300 GOODFELLOW BOULEVARD ST. LOUIS, MO 63120
12. REPORT DATE: JANUARY 1980
13. NUMBER OF PAGES:
14. MONITORING AGENCY NAME & ADDRESS (if different from Controlling Office):
15. SECURITY CLASS. (of this report): UNCLASSIFIED
15a. DECLASSIFICATION/DOWNGRADING SCHEDULE:
16. DISTRIBUTION STATEMENT (of this Report)

Approved for public release; distribution unlimited.

17. DISTRIBUTION STATEMENT (of the abstract entered in Block 20, if different from Report)

18. SUPPLEMENTARY NOTES

19. KEY WORDS (Continue on reverse side if necessary and identify by block number)

Automatic Stabilator	Digital Stability Augmentation System
Engineer Design Test 4	Field of View
Handling Qualities	Hover Performance
Level Flight Performance	Low-speed Flight
Nap of the Earth	Vibration

20. ABSTRACT (Continue on reverse side if necessary and identify by block number)

Engineer Design Test 4 (EDT-4) was conducted using a prototype YAH-64 aircraft (S/N 77-23258). This test was conducted at Palomar Airport, Carlsbad, California, (elevation 328 feet) between 10 and 29 November 1980. Approximately 33 hours were flown during 27 flights. Major changes affecting performance and handling qualities were made to the YAH-64 since the last evaluation (EDT-2). These included a new, digital stability augmentation system, and a redesigned empennage featuring an automatically programmed stabilator and an increased diameter tail rotor. In previous evaluations the flight envelope and scope of test were limited by structural loads and continuous monitoring of those loads was required for all flights. During this test the flight envelope was much larger and no structural limits were encountered. Hover and level flight performance of the aircraft have been improved since

EDT-2. Handling qualities have been greatly improved. The most significant areas of handling qualities improvements were in low-speed flight characteristics, including directional control margins, short-period dynamic stability characteristics (particularly in high speed flight), and in aircraft pitch attitude during approaches and IRP climbs. Three deficiencies were found during this evaluation: The disengagement of the HARS, DASE, and automatic operation of the stabilator, and erroneous activation of the engine out/low rotor speed audio warning tone with failure of the No. 1 generator; random failure of the master caution light to illuminate with the illumination of some caution or warning panel segment lights; and the restricted pilot's field of view caused by canopy frame structure during NOE and contour flight. Vibration levels have been significantly reduced since EDT-2 but are still objectionable. Forty-four shortcomings were found.

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SUBJECT: Directorate for Development and Qualification Position on the Final Report of USAAEFA Project No. 80-03, Engineer Design Test 4, YAH-64 Advanced Attack Helicopter

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1. The purpose of this letter is to establish the Directorate for Development and Qualification position on the subject report. This test was conducted using prototype YAH-64, S/N 77-23258 for the purposes of evaluating improvements in the helicopter. The objectives were to assess the flight handling characteristics with a new empennage configuration, evaluate flight performance, assess changes incorporated to improve vibration characteristics and provide data in establishing the Airworthiness Release for EDT-5.

2. This Directorate agrees with the report findings and conclusions with the following exceptions, and are directed to the report paragraph as indicated.

a. Paragraph 85e. The revised directional control stop tested is not the final production yaw actuator stop setting. This setting will be established at the conclusion of the altitude testing and be consistent with the specification requirements for directional control margins. A reduced actuator stop setting (equivalent to approximately 27^{o} blade angle) will be used for those aircraft participating in OT-II, however, for structural reasons.

b. Paragraph 85f. We disagree with the statement that vibration levels were not significantly affected by the removal of the vertical vibration absorber. The absorber was removed midway through EDT-4 for inspection and reinstalled improperly. Consequently, the last half of the flight tests did not generate representative absorber installed data. However, this data was used for comparison with the absorber removed vibration levels. Additionally, a comparison of the low speed flight test results generated prior to removal and reinstallation of the absorber show a significant improvement. This is also supported by the Hughes Helicopters (HH) test results obtained prior to the start of EDT-4. The most important point, however, is the recommendation for further improvement in vibration levels (see paragraph 90c) with which this Directorate strongly supports.

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c. Paragraph 86a. As a result of the reported deficiency, HH has modified the electrical system to prevent deactivation of critical relays and incorporated a third buss for OT-II aircraft which will prevent disengagement of the HARS, DASE and automatic operation of the stabilator, and erroneous activation of the engine out/low rotor speed audio tone in the event of a generator failure.

d. Paragraph 86b. The random failure of the master caution light to illuminate with the illumination of some caution or warning panel segment lights has been traced to a faulty master caution panel unit. This unit was replaced and the same failure reoccurred. There appears to be a problem with the caution panel segments not making proper contact to illuminate the master caution light. HH should have this corrected prior to the start of OT-II.

e. Paragraph 86c. The reported restricted pilot's field of view caused by canopy frame structure during NOE and contour flight should not be considered a deficiency only on the basis of an engineering flight evaluation. It is important that qualitative assessments of the aircraft be observed and reported, but the degree of acceptability should remain with the operational evaluators. The recommendation in paragraph 90b suggests emphasis be placed on the characteristics during future operational tests. To increase the probability of successful operational evaluation, the stabilator system has been changed so that pilots can use the stabilator in the manual mode for more efficient attitudes during NOE and contour flying and the maximum down position during manual operation has been increased to 45^{o}.

f. Paragraph 87b. The reported poor design of the pilot's fuel control panel will be corrected for the production aircraft. A magnetic latching switch for the Boost Pump and Tank Select switches will be incorporated so that boost pump ON operation cannot be initiated unless Tank Select is in AFT TK. Likewise, if Tank Select is moved to any position from AFT TK, the boost pump switch will automatically go to OFF. The boost pump ON light will be moved to the caution/advisory panel and a transfer pump function light on the caution/advisory panel will be added. The corrections to the fuel panel will provide an adequate fuel system for the production of AAH.

g. Paragraph 87c. Reported HARS inaccuracies reflected excessive alignment times and heading errors once aligned. The current alignment times exceed specification requirements, but should be corrected for production. The HARS accuracy has been demonstrated to be within specification requirements, and is satisfactory for OT-II testing. The test unit was apparently malfunctioning and should be examined.

h. Paragraph 87d. The absence of SAS pitch rate damping at load factors greater than 1.6 due to saturation of the pitch SAS actuator is not considered a significant enough shortcoming, as indicated in paragraph 87, to warrant modification to the DASE.

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i. Paragraph 87f. The reported objectionable 4/rev vertical vibration in rearward flight will be reduced with the use of manual stabilator incidence scheduling. The reported high lateral vibration levels in right sideward flight and high vertical vibration levels during the termination of an approach are characteristic of the helicopter but can be reduced with properly tuned vibration absorbers.

j. Paragraph 87h. Although the poor trimmability at airspeeds between 85 and 110 KCAS is reported as a shortcoming, it is important to note that the IMC evaluation with DASE on yielded a HQRS of 3-4, thereby implying that trimmability appears to be an insignificant problem. Additionally, if the aircraft is enroute at this airspeed range, which is slower than a normal "enroute" speed, and maintaining level flight, then the attitude hold mode engagement would minimize excursions, thereby reducing pilot workload. Therefore, no corrective action is planned at this time.

k. Paragraph 87j. The final determination of the IMC capability should be held until an evaluation of the helicopter with production flight instruments is performed. DASE modifications can possibly be implemented to minimize sideslip excursions from a referenced value equal to "ball centered" (i.e., zero bank angle) flight as opposed to a zero sideslip reference.

l. Paragraph 87l. Although the longitudinal control trim shift from 12 to 18 KTAS is reported as a shortcoming, it is not considered significant enough to warrant modification to the airframe/flight control system.

m. Paragraph 87m. The degree of control trim shifts between 16 and 24 KTAS rearward and between 13 and 18 KTAS forward are not believed to warrant modification to the airframe/flight controls.

n. Paragraph 87t. The improper operation of the stabilator during minimum power descents at 50 and 60 KCAS has been corrected. The random dropout was resolved by increasing the allowable difference between pitot 1 and pitot 2, thereby decreasing the effects of airspeed perterbations. Inaccuracies in the ADS were corrected, thereby improving the stabilator positioning below 60 knots (ADS provides the airspeed signal to the stabilator system below 60 knots).

o. Paragraph 87u. The periodic sampling of hydraulic fluid, as required by Hughes Program Directive 077, is being conducted to check against contamination and to provide baseline data for wear rate to be used in the establishment of preventive maintenance procedures. Production aircraft will not require periodic sampling of flight control hydraulics and therefore cost need not be incurred to provide an easier method for sampling.

p. Paragraph 88b. The system tested operated per the specification which provides light segments for each 2% of range and was previously approved at a lighted mock-up inspection by the Army. Any action to display the full green range during normal operation will be held in abeyance until the completion of operational testing.

q. Paragraph 88f. The location of the pilot engine control quadrant is not considered a shortcoming. The layout of this quadrant was carefully evaluated at design reviews, mock-up reviews and previous Government flight testing.

r. Paragraph 88j. The constant illumination of the lower green segment light on the Marconi vertical scale was designed into the system to provide the pilot an indication that electrical power is being supplied to the instrument and will be retained.

s. Paragraph 88l. The anthropometric design of the pilot's cyclic grip is not considered a shortcoming. The design of the grip meets the 5th-95th percentile operator. However, this grip is not considered satisfactory for the total system's operation of the AAH and a new standard grip is available for the production aircraft.

t. Paragraph 88n. The cockpit, seats and controls were designed to accommodate the 5th and 95th percentile aviator using US Army anthropometric measurements. As a cost avoidance item, longitudinal seat adjustments were not included in the AAH specification.

u. Paragraph 88o. The problem in operating the HARS control switch is due to the smooth knob installed on the test aircraft. The production design will incorporate a knurled knob for easy activation.

v. Paragraph 88r. All switches were allocated on a priority basis and reviewed during design, mock-up and cockpit reviews. A better evaluation of the adequacy of these switches will be determined during the icing survey.

3. As relates to recommendations, this Directorate concurs, with the exception of paragraph 90d (relating to shortcoming 87d), 90o and 90q. Rationale for these exceptions are apparent in the appropriate comments of paragraph 2 above.

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4. As noted in the report, AEFA conducted those tests necessary to measure power required in level flight and for out-of-ground effect hover. This data (corrected for test instrumentation drag effects) must be used in conjunction with power available to calculate cruise performance and rate of climb. Using calculated power available from the T700-GE-700 model specification and the installation losses measured during contractor flight test, it can be determined that the maximum mission gross weight that the YAH-64 will meet the 450 ft per minute vertical rate of climb requirement using 95% of Intermediate Rated Power when operating under 4000 ft/35°C temperature conditions is approximately 14,325 pounds. At this mission gross weight, the cruise speed at Maximum Continuous Power is approximately 140 KTAS, which is less than the Military Need requirement of 145 knots. Incorporation of the T700-GE-701 engine currently planned by the Army Program Manager will result in cruise speed performance which will meet the Military Need and vertical rates of climb substantially greater than the 450 ft requirement without a major weight reduction effort for the production model.

5. The overall results of this evaluation substantiate the modified empennage design (stabilator) and supports the continued evaluation of the YAH-64 in the attack helicopter role.

FOR THE COMMANDER:

Charles C. Crawford

CHARLES C. CRAWFORD, JR.
Director of Development
and Qualification

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INTRODUCTION

BACKGROUND

1. In June 1973, the United States Army Aviation Systems Command (since renamed the United States Army Aviation Research and Development Command), awarded a Phase 1 Advanced Development Contract to Hughes Helicopters (HH). The contract required HH to design, develop, fabricate, and initiate a development/qualification effort on two Advanced Attack Helicopter (AAH) prototypes and a ground test vehicle as part of a Government Competitive Test. The United States Army Aviation Engineering Flight Activity (USAAEFA) conducted Development Test 1 (DT-1) using two of these aircraft (ref 1, app A). In December 1976, the United States Army Aviation Research and Development Command (AVRADCOM) awarded a Phase 2 Engineering Development Contract to HH for further development and qualification of the YAH-64 to include subsystems and mission equipment. During this program, Engineer Design Tests (EDT) 1 and 2 were conducted to evaluate developmental progress (refs 2 and 3). In August 1980, AVRADCOM requested USAAEFA to conduct this test, EDT-4, which follows major changes to improve flying qualities, structural integrity, flight performance and vibration characteristics (ref 4). A test plan (ref 5) was submitted in September 1980, and an Airworthiness Release (ref 6) was issued in October 1980 and revised in November 1980.

TEST OBJECTIVES

2. The objectives of this test were to assess the flight handling characteristics with the new empennage configuration, evaluate flight performance parameters, evaluate changes made to improve vibration characteristics, and provide data to be used in establishing the air vehicle flight envelope for EDT-5.

DESCRIPTION

3. The YAH-64 is a two-place, tandem-seat, twin-engine helicopter with four-bladed main and antitorque rotors and conventional wheel landing gear. The helicopter is powered by two General Electric YT700-GE-700R turboshaft engines. The YAH-64 has a moveable horizontal stabilator. A 30mm gun is mounted on the underside of the fuselage below the front cockpit. The helicopter has a wing with two store pylons on each side to carry HELLFIRE missiles or 2.75-inch folding fin aerial rockets. The test aircraft was US Army serial number (S/N) 77-23258. Major changes to the helicopter since EDT-2 include:

a. Moveable horizontal stabilator

b. Increased tail rotor diameter

c. Digital Automatic Stabilization Equipment (DASE)

d. Wing flap position fixed

e. Drag reduction fairings

f. Vibration absorbers

g. Stiffened canopy.

An aerodynamic mockup of the Martin-Marietta Target Acquisition Designation System/Pilot Night Vision System (TADS/PNVS) was installed. Further description of the helicopter is contained in the System Specification (ref 7, app A) and in appendix B.

TEST SCOPE

4. Flight testing for EDT-4 was conducted at Palomar Airport, Carlsbad, California (328-foot elevation) from 3 through 19 November 1980. A total of 27 flights were conducted during which 32.7 hours were flown. Two Army pilots performed the evaluation with an HH pilot acting as the aircraft commander. HH installed, calibrated, and maintained the test instrumentation and performed all aircraft maintenance during the test. Flight restrictions and operating limitations contained in the Airworthiness Release issued by AVRADCOM and the Draft Contractor's Flight Manual (ref 8, app A), were observed during the evaluation. Where possible, flight test data were compared to the system specification and results obtained during EDT-2. The scope of the test is shown in table 1.

TEST METHODOLOGY

5. Established flight test techniques and data reduction procedures were used (refs 9 and 10, app A). Test methods are briefly discussed in the Results and Discussion section of this report. A vibration rating scale (VRS) (fig 1, app D) was used to augment crew comments relative to aircraft vibration levels. A handling qualities rating scale (HQRS) (fig 3, app D) was used to supplement pilot comments on the handling qualities. Flight test data were obtained from calibrated test instrumentation and were recorded on magnetic tape. Real time telemetry was used to monitor selected critical parameters throughout the flight test. A detailed listing of the test instrumentation is contained in appendix C. Data analysis methods are described in appendix D.

Table 1. Test Scope[1]

Type of test	Gross Weight (lb)	Longitudinal Center of Gravity Location (fs)	Density Altitude (ft)	Trim Calibrated Airspeed (kt)	Remarks
Hover performance	13000 to 17960	205.0 (Aft)	1220	0	100-foot wheel height
Level flight performance	14600	202.0 (Fwd)	1360	43 to 147	Thrust coefficient, C_T = 0.006651
	14860	202.0 (Fwd)	5420	43 to 143	C_T = 0.007644
	14740	202.0 (Fwd)	6380	43 to 138	C_T = 0.008337
	15640	202.2 (Fwd)	9720	43 to 123	C_T = 0.009187
Control positions in trimmed forward flight	15200	202.2 (Fwd)	4940	51 167	Level flight and dives at IRP[2]
	15600	202.3 (Fwd)	5700	49-122	Climbs at IRP[2]
	15580	202.3 (Fwd)	4470	50-124	Minimum power descents
Collective-fixed static longitudinal stability	14260	205.8 (Aft)	5200	53	
	14509	205.9 (Aft)	5200	147	
Static lateral-directional stability	14500	205.7 (Aft)	4400	56	
	14740	205.8 (Aft)	5440	144	
Maneuvering stability	14800	205.9 (Aft)	5440	145	Right and left turns, pullups and pushover
Dynamic stability[3]	15040	206.5 (Aft)	5620	70 to 145	Short period, attitude hold on and off
	14340	205.9 (Aft)	4660	70, 145	Long period, attitude hold on and off
Low-speed flight characteristics[3,4]	14720	201.4 (Fwd)	280	-40 to 50	Rearward and forward DASE ON
	14820	201.3 (Fwd)	440	46 Lt to 49 Rt	Sideward flight DASE ON
	14800	201.9 (Fwd)	-230	43 Lt to 47 Rt	Sideward flight DASE OFF
	14620	201.5 (Fwd)	280	30	Critical azimuth determination DASE ON
	14520	201.7 (Fwd)	20	-3 to 49	Critical azimuth flight DASE ON
	14500	201.9 (Fwd)	-180	-2 to 43	Critical azimuth flight DASE OFF
Mission maneuvering characteristics[3]	14300 to 16200	Fwd and Aft	0 to 6000	-45 to V^5_{NE}	Accelerations/decelerations, approaches, nap of the earth, contour simulated IMC[6] flight
Simulated engine failures	14480	205.7 (Aft)	3740	76	IRP[2] climb
	14360	206.7 (Aft)	4640	144	Level flight at IRP[2]
	14340	206.7 (Aft)	3780	82	Minimum power descent
Simulated DASE failures	14680	201.8 (Fwd)	1140	142	Disengagements in level flight
Simulated stabilator failures	14780	202.1 (Fwd)	100	0 - 100	Accelerations, stabilator fixed at 24 degrees
				140 - 0	Approaches, stabilator fixed at -3 degrees
	13940	204.9 (Aft)	5400	60 to 122	Hardover failures

NOTES:

[1] Configuration 8-HELLFIRE except hover performance which was flown clean wing. Rotor speed 290 RPM (100%) except hover performance which was flown at 281, 290, and 298 RPM, DASE ON.
[2] IRP = Intermediate Rated Power
[3] Digital Automatic Stabilization Equipment (DASE) ON and OFF
[4] Airspeeds listed are true rather than calibrated.
[5] V_{NE} = never exceed airspeed
[6] IMC = Instrument Meteorological Conditions

RESULTS AND DISCUSSION

GENERAL

6. Major changes affecting performance and handling qualities were made to the YAH-64 since the last evaluation (EDT-2). These included a new, digital stability augmentation system, and a redesigned empennage featuring an automatically programmed stabilator and an increased diameter tail rotor. In previous evaluations the flight envelope and scope of test were limited by structural loads and continuous monitoring of those loads was required for all flights. During this test the flight envelope was much larger and no structural limits were encountered. Hover and level flight performance of the aircraft have been improved since EDT-2. Handling qualities have been greatly improved. The most significant areas of handling qualities improvements were in low-speed flight characteristics, including directional control margins, short-period dynamic stability characteristics (particularly in high speed flight), and in aircraft pitch attitude during approaches and IRP climbs. Three deficiencies were found during this evaluation: The disengagement of the HARS, DASE, and automatic operation of the stabilator, and erroneous activation of the engine out/low rotor speed audio warning tone with failure of the No. 1 generator; random failure of the master caution light to illuminate with the illumination of some caution or warning panel segment lights; and the restricted pilot's field of view caused by canopy frame structure during nap-of-the-earth (NOE) and contour flight. Vibration levels have been significantly reduced since EDT-2 but are still objectionable. Forty-four shortcomings were found.

PERFORMANCE

General

7. Out-of-ground-effect (OGE) hover performance and level flight performance were measured during this program. Power required for hovering and level flight is less for the EDT-4 aircraft than for the EDT-2 aircraft. Engine power losses due to installation on the airframe were not determined during this test. Therefore, the performance requirements of the system specification could not be checked. Performance tests were flown at the conditions listed in table 1.

Hover Performance

8. Hover performance was measured at a 100-foot wheel height using primarily the tethered technique with a few free-flight data points. Figure 1, appendix E, presents nondimensional hover performance of the YAH-64 as measured during both EDT-4 and EDT-2. Each of the three lines labeled EDT-2 represent a different rotor speed. Tail rotor efficiency changed significantly with rotor speed and caused the separate performance curves. This was manifested as a change in tail rotor power required with a change in rotor speed at constant main rotor thrust coefficient. During EDT-4, nondimensional hover performance was not affected by changing rotor speed. At a gross weight of 14,200 pounds and atmospheric conditions of 4,000 feet, pressure altitude, and 35°C, the EDT-4 aircraft required 2083 shaft horsepower (SHP) to hover OGE at 100 percent rotor speed. This represents a reduction of 119 SHP (5.4 percent) from the EDT-2 configuration at the same conditions.

Level Flight Performance

9. Power required for level flight was measured in the 8-HELLFIRE configuration at a constant rotor speed (290 rpm). Nondimensional data are presented in figures 2 through 4, appendix E. Figure 5 presents a comparison of performance measured during EDT-4 and EDT-2. Dimensional data for each flight are presented in figures 6 through 9. At the conditions shown in figure 5, the power required for level flight at 145 knots, true airspeed (KTAS) has been reduced from 2090 SHP in EDT-2 to 2065 SHP in EDT-4.

HANDLING QUALITIES

General

10. The handling qualities of the YAH-64 were evaluated at the test conditions listed in table 1. All tests were conducted using standard flight test techniques (ref 10). Maneuvers were flown at zero sideslip where possible. All tests with the exception of longitudinal control system characteristics were conducted with the copilot/gunner (CPG) cyclic stick in the retracted position. Test results were compared, where possible with those previously reported (refs 1 through 3, app A). Additionally, the test data were compared with the requirements of the YAH-64 Phase 2 System Specification (ref 7, app A), where possible.

Control System Characteristics

11. The control system mechanical characteristics were evaluated on the ground with external hydraulic and electrical power applied to the aircraft and the rotors stopped. All measurements were taken at pilots station. Tests were performed with the trim feel system ON and OFF. Longitudinal control system characteristics were evaluated with the CPG cyclic stick extended and retracted which affected the mass balance of the control system. Results were qualitatively verified in flight. Data are presented in figures 10 through 17, appendix E. Table 2 is a summary of the control system mechanical characteristics observed during these tests. The control system characteristics have generally been improved over those previously reported during EDT-2 (ref 3, app A).

12. The longitudinal control system characteristics showed a significant improvement in control centering over that previously reported in EDT-2. With trim feel ON control centering was positive with a trim control displacement band of 0.2 inches which is satisfactory. With the trim feel OFF control force gradients were essentially zero and control centering was absent. Control forces could not be trimmed to zero in flight with the CPG cyclic stick extended and slight forward pressure on the cyclic control was required to maintain the desired trim position. The longitudinal breakout force (plus friction) was excessive. The 2.5 pound breakout force contributed to increased pilot workload when attempting to maintain a precise hover (HQRS 4) and may be significant when hovering with reference to PNVS symbology. The excessive longitudinal breakout force (plus friction) is a shortcoming previously reported. The longitudinal control system characteristics failed to meet requirements of the following paragraphs of reference 7, appendix A.

Table 2. Control System Mechanical Characteristics[1]

Test Parameter	Control System			
	Longitudinal[2]	Longitudinal[3]	Lateral	Directional
Breakout force (plus friction) (lb)	1.5 Fwd, 2.5 Aft	2.5 Fwd, 1.4 Aft	1.5 Left, 1.4 Right	5.0 Left, 5.0 Right
Full control travel (in)	10.2	10.2	8.9	5.4[4]
Control oscillation	None	None	None	None
Free play (in)	Negligible	Negligible	Negligible	Negligible
Mechanical coupling	None	None	None	None
Force to move stick 0.5 inch from trim (lb)	2.0 Fwd, 2.8 Aft	3.1 Fwd, 1.8 Aft	1.9 Left, 2.0 Right	N/A
Limit control force (lb)	8.2 Fwd, 9.0 Aft	8.2 Fwd, 9.0 Aft	8.5 Left, 9.7 Right	20.0 Left, 19.0 Right
Control centering	Positive	Positive	Positive, Absolute	Positive
Control jump	Negligible	Negligible	Negligible	Negligible
Control forces trimmable to zero	Yes	No	Yes	Yes
Force gradient (lb/in)	0.9 Fwd, 0.5 Aft	0.6 Fwd, 0.6 Aft	0.8 Left, 0.9 Right	2.5 Left, 2.4 Right

NOTES:
[1] Rotors static, ground hydraulic and electrical power applied.
[2] Copilot-gunner cyclic control stick retracted.
[3] Copilot-gunner cyclic control stick extended.
[4] Revised directional control stop reduced full control travel to 4.2 inches.

10.3.2.1.1 - the longitudinal breakout force (plus friction) exceeded the 1.5 pound limit by 1.0 pound.

10.3.2.5 - the longitudinal control force could not be trimmed to zero in flight with the CPG cyclic stick extended.

13. Lateral control characteristics also showed an improvement compared to EDT-2 in control centering (trim feel ON) with positive, absolute centering observed. Control force gradients were essentially the same as those for the longitudinal control and though not in compliance with specification requirements were satisfactory in flight. Breakout force (plus friction) was symmetrical about trim and there was no tendency to develop any significant pilot induced oscillation (PIO) laterally. Trim feel OFF operation was essentially the same as that observed for the longitudinal control with no force gradients and no control centering. The lateral control characteristics are satisfactory although the lateral control force gradient failed to meet the requirements of paragraph 10.3.2.2 of reference 7, appendix A in that the maximum of 60 percent of the longitudinal force gradient was exceeded.

14. The directional control characteristics were improved in that breakout force (plus friction) had been reduced from a maximum of 9.5 pounds, reported in EDT-2 to a maximum of 5.0 pounds. Undesirable control jump when retrimming (a previously reported shortcoming) was not observed during this test. Directional control centering was less positive than previously reported but posed no particular problem in flight. The directional control system characteristics are satisfactory.

Control Positions in Trimmed Forward Flight

15. The control positions in trimmed level flight and dives were evaluated from 51 to 167 knots, calibrated airspeed (KCAS). Data are presented in figure 18, appendix E. The aircraft exhibited an undesirable variation in the trimmed longitudinal control position between 85 and 110 KCAS. Within this airspeed range less than 1/4 inch of cyclic control position change was required. This resulted in poor trimability and will contribute significantly to pilot workload during instrument meteorological conditions (IMC) flight. The poor trimmability at airspeeds between 85 and 110 KCAS is a shortcoming previously reported.

16. Control positions during intermediate rated power (IRP) climbs and minimum power descents (less than 10 percent torque per engine) were evaluated at airspeeds between 49 and 124 KCAS. Data are presented in figure 19, appendix E. The pitch attitude variation with power application has been greatly improved. The field of view during IRP climbs, a previously reported shortcoming, is now satisfactory. The longitudinal control trim shift with power changes has also been significantly improved. A trim shift observed during this test was 1 inch and is satisfactory.

17. Failure of automatic stabilator operation was observed during minimum power descents at 50 and 60 KCAS. The stabilator ran away to positions considerably off the design schedule and then shut down the automatic mode. Re-engagement required only pushing the reset button. The control positions are shown in figure 19, appendix E. The maximum stabilator deflection attained was 23 degress trailing edge down. This required a longitudinal control position of 6.9 inches from full forward. The remaining control position margin of 3.3 inches was adequate and aircraft control was maintained throughout the test. The failure appeared to have been caused by erroneous airspeed information supplied to the stabilator control

unit. The improper operation of the automatic stabilator during mimimum power descents at 50 and 60 KCAS is a shortcoming.

18. Yaw oscillations of ±3 degrees were observed at airspeeds between 40 and 70 KCAS. The oscillations were annoying to the pilot and appeared to be caused by the digital automatic stabilization equipment (DASE) attempting to maintain flight at zero sideslip. The pilot was unable to damp the oscillations with directional inputs. The oscillations did not appear to be a problem during NOE flight. Sideslip information was provided to the DASE computer from a transducer installed on a test instrumentation boom rather than from the Air Data Sensor (ADS) as proposed for the production configuration. The response of the transducer may have contributed to the yaw oscillation observed. The yaw oscillations at airspeeds between 40 and 70 KCAS should be re-evaluated with the ADS providing sideslip information to the DASE.

19. The attitude hold mode incorporated in the DASE was evaluated during trimmed level flight. With attitude hold engaged, pilot effort required to maintain airspeed was reduced. When retrimming (depressing the trim release button on the cyclic grip) with the attitude hold engaged uncommanded aircraft response was noted in all three axes. The response was greatest in yaw (±10 degrees) and was more pronounced when retrimming after flying with the attitude hold on for one minute or more. The aircraft response was annoying and contributed to increased pilot workload when attempting to establish a precise trim condition. The uncommanded aircraft response when retrimming with the attitude hold engaged is a shortcoming. Consideration should be given to incorporating an altitude hold feature in the attitude hold mode of the DASE.

Static Longitudinal Stability

20. The static longitudinal stability characteristics were evaluated at trim airspeeds of 56 and 147 KCAS. The data from these tests are provided in figures 20 and 21, appendix E. The longitudinal cyclic control position gradient indicates approximately neutral stability at both airspeeds tested. The lack of positive static longitudinal stability, although satisfactory and desirable for the low speed attack mission, will contibute to increased pilot workload during IMC flight at higher airspeeds where more precise airspeed control is required (para 47).

Static Lateral-Directional Stability

21. The static lateral-directional stability characteristics were evaluated at the condition's shown in table 1. The aircraft was trimmed in level flight at zero sideslip. The collective control position was approximately constant and sideslip angles were varied in 5-degree increments (left and right) while maintaining constant airspeed. Data are presented in figures 22 and 23, appendix E. At 56 KCAS the helicopter exhibited positive directional stability (increased left directional control for increase in right sideslip), and positive dihedral effect (increased right lateral control with increased right sideslip). Sideforce cues were weak about trim at this airspeed as evidenced by the small change in roll attitude. At 144 KCAS directional stability and dihedral effect were again positive however, sideforce cues were significantly increased. At zero sideslip the left roll attitude was approximately 7 degrees. This attitude was uncomfortable to the pilot and may be a significant problem during IMC flight because the DASE attempts to maintain zero sideslip at airspeeds greater than 50 knots (para 47). Except for the roll attitude at zero sideslip, the static lateral-directional stability characteristics are satisfactory.

Maneuvering Stability

22. The maneuvering stability characteristics were evaluated using constant airspeed left and right turns and symmetrical pullups and pushovers at the flight conditions listed in table 1. The stick-fixed stability (control position vs load factor) was positive up to load factors of 1.6 (fig 24, app E). Above 1.6g the stability became neutral and pitch rate damping was lost due to saturation of the pitch stability augmentation system (SAS) actuator. A time history of pitch SAS actuator position is shown in figure 25, appendix E. Random, uncommanded pitch excursions were observed during steady turns at bank angles of greater than 40 degrees. The loss of pitch rate damping was also observed during contour flight when making turns at 1.6g and greater. This was annoying to the pilot as the aircraft exhibited a pitch up or "dig in" tendency which significantly increased pilot work load (HQRS 5). The absence of SAS pitch rate damping at load factors greater than 1.6 due to the saturation of pitch SAS actuator is a shortcoming which should be corrected prior to operational testing.

Dynamic Stability

23. The short-term dynamic stability characteristics of the YAH-64 were evaluated at the conditions listed in table 1. Aircraft motions were induced by one inch lateral, longitudinal, and directional doublets, with the DASE both ON and OFF and attitude hold OFF. Following the input all controls were held fixed until the motion subsided or until recovery became necessary. A typical time history of the short-term aircraft response, DASE OFF, is presented in figure 26.

24. The response of the YAH-64 to control doublets with the DASE ON was essentially deadbeat and is satisfactory. The response of the YAH-64 to DASE OFF control doublets was a short-term three-axis oscillation. This coupled three-axis oscillation (with 4-second period) was excited by control inputs in any of the three axes and was similar in character for all of the inputs. The damping of the oscillation decreased with increasing airspeed until the short-term stability became neutral at 115 KCAS. At airspeeds greater than 115 KCAS, the stability remained neutral. The trend found during EDT-2 of a large decrease in stability with an increase in airspeed was not present in EDT-4. The oscillations were controllable by the pilot flying in visual meterological conditions (VMC) with some increase in pilot work load. However, the pilot flying in IMC may encounter a significant increase in pilot work load when flying in turbulent conditions DASE OFF (para 47).

25. The longitudinal long-term dynamic stability characteristics of the YAH-64 were evaluated at the conditions in table 1. Aircraft motions were introduced by displacing the cyclic control aft of trim, allowing the airspeed to decrease by 10 KCAS and returning the control to trim. All controls were then held fixed until the aircraft motions subsided or until a limit condition was reached. These tests were performed DASE ON and OFF and attitude hold ON and OFF. A typical time history of the aircraft response to a ten knot slow excitation at 80 KCAS with the DASE ON and attitude hold OFF is presented in figure 27.

26. The aircraft longitudinal long-term response with the attitude hold ON was essentially deadbeat and is satisfactory. The aircraft response with the attitude hold OFF and with the DASE ON or OFF at 80 KCAS was divergent. The period of the oscillation was approximately 70 seconds. The pilot could control the oscillation; however, the ability of the pilot to maintain precise pitch control and airspeed control while flying IMC will be degraded.

Ground Handling Characteristics

27. The ground handling characteristics of the YAH-64 with DASE ON and OFF were evaluated throughout this test program on both concrete and macadam taxiways. Wind conditions were generally less than 25 knots. During this evaluation the yaw DASE authority was 10 percent and the Command Augmentation System (CAS) was automatically disengaged by a touchdown squat switch located on the left main landing gear. Directional control while taxiing was easily accomplished with the DASE ON. With the DASE OFF (no yaw rate damping) the pilot tended initially to overcontrol directionally; however, within a short time the pilot could easily compensate for the lack of yaw rate damping.

28. The YAH-64 requires a high brake pedal pressure during ground taxi operations and when setting and releasing the parking brake. In addition to the high pedal pressure, the design of the pedals requires the pilot to raise his heels off the floor to apply brake pedal pressure. Additionally, a portion of the cockpit structure above the pedals occasionally restricted the pilot from removing his feet from the brake pedals after a brake application. The above factors combined to cause inadvertent directional control inputs during brake application which is a shortcoming previously reported.

29. The tail wheel lock/unlock light is located on the upper left of the pilot's instrument panel behind the canopy jettison handle and below the glare shield. Landings and takeoffs are performed with the tailwheel locked and ground taxiing is performed with the tailwheel unlocked. After landing or before takeoff, the pilot must activate the tailwheel lock/unlock switch and then watch for the illumination of the tailwheel unlock light. This process requires that the pilot divert his attention from outside the cockpit to the tailwheeel unlock light The poor location of the tailwheel unlock light is a shortcoming previously reported.

30. The parking brake handle position is the only cockpit indication of whether the brakes are set or released. During ground operations, the brakes were found to be set with the parking brake handle not in the full out (set) position. The lack of a reliable indication of parking brake status is a shortcoming previously reported.

Takeoff and Landing Characteristics

31. Takeoffs and landings were evaluated during each flight. Normal, maximum performance, minimum power, running, and level acceleration takeoffs were accomplished. Normal, steep approach, running and simulated autorotative landings were accomplished. Figure 28 is a time history of a normal takeoff. The longitudinal control displacement from a hover to 70 KCAS was approximately 1 inch. On previous evaluations (EDT-2) the control displacement was 2 inches. The longitudinal control displacement required during takeoff is now considered satisfactory.

32. Figures 29 and 30 are typical time histories of a simulated autorotation and a steep approach, respectively. The pitch attitudes required during these maneuvers are improved from EDT-2, however, the point of intended landing was blocked from the pilot's view at a height above the ground of 150 to 200 feet. The primary restrictions to the field of view during approach were the PNVS turret (located on the nose of the aircraft) and the CPG helmet which obstructed approximately 40 percent of the forward windscreen. The restricted pilot's forward field of view during a steep approach and a simulated autorotation (previously reported as a

deficiency) is now considered a shortcoming which should be identified as an area of special interest during operational testing.

Low-Speed Flight Characteristics

33. The low-speed flight characteristics of the YAH-64 were evaluated using calibrated ground pace vehicle as a speed reference. Surface wind conditions were less than 5 knots. The test aircraft wheel height was approximately 20 feet in-ground-effect (IGE). Flights were conducted with the DASE ON and OFF. Data are presented in Figures 31 through 38.

Forward and Rearward Flight:

34. Figure 31 shows the control positions in low-speed forward and rearward flight. The longitudinal control position variation with airspeed is nonlinear but satisfactory. Large, abrupt lateral control trim shifts occur between 16 and 24 KTAS rearward and between 13 and 18 KTAS forward airspeeds. When executing hovering flight and hovering turns in gusty wind conditions, the pilot will not be able to maintain a precise ground track. The large, abrupt control trim shifts between 16 and 24 KTAS rearward and between 13 and 18 KTAS forward flight is a shortcoming.

Sideward Flight:

35. In general, the trends of lateral cyclic and directional controls with airspeed were proper. Minor reversals were noted but were not objectionable. During the conduct of this evaluation and following the low speed flight tests, the left tail rotor stop was changed from a nominal 33 degrees to 27 degrees. With the 33 degrees rigging there was approximately 1.8 inches of left directional control margin at 45 KTAS right sideward flight. However, with the 27 degrees rigging only 0.6 inch would have remained. Both riggings reflected an improved directional control margin since EDT-2 for test day conditions. At higher gross weights and density altitudes, the directional control margin with 27 degrees rigging may be inadequate. Figure 32 also shows the variation of longitudinal control with lateral velocity. There is an approximate 1 inch longitudinal trim shift between 10 KTAS and 25 KTAS left sideward flight. This longitudinal trim shift makes precise pitch control and maintenance of a precise ground track difficult during acceleration from 12 to 18 KTAS left sideward flight. The abrupt longitudinal control trim shift from 12 to 18 KTAS left sideward flight is a shortcoming. A reevaluation of the revised tail rotor rigging from 33 degrees to 27 degrees should be conducted.

36. Trim control positions and pilot work load bands are presented in Figure 33 for DASE OFF sideward flight. The highest work load was at speeds greater than 15 KTAS left sideward flight. The pilot was able to control heading within ±3 degrees to 45 KTAS left and right sideward flight with an increase in longitudinal and directional control work load. The ease of control in DASE OFF sideward flight is a significant improvement from EDT-2.

Critical Azimuth Flight:

37. A flight was performed to determine the critical azimuth. The critical azimuth was found to be approximately 240 degrees relative to the nose of the aircraft in a clockwise direction (fig 34). Critical azimuth was defined by high pilot work load

since all control margins were adequate. Figures 34 and 36 show the control positions and work load bands for flight at the critical azimuth DASE ON and OFF, respectively. The pilot was able to control aircraft heading within ±4 degrees at the critical azimuth, DASE ON and OFF to 45 KTAS. This represents a significant improvement from EDT-2 and is satisfactory (aircraft directional control was lost during EDT-2).

Low-Speed Maneuvering:

38. Directional control step inputs were made while in left and right sideward flight at 35 KTAS with the DASE ON to check specification compliance. Data for a left directional control input is presented in figure 37. In both left and right sideward flight, yaw rates of greater than 15 deg/sec were generated within 1.5 seconds which meets the systems specification requirement. These tests were performed with the 33 degrees left pedal stop tail rotor rigging. It is projected that specification requirements would also have been met, for test day conditions, with the revised tail rotor rigging of 27 degrees.

39. Lateral accelerations from a hover to 35 KTAS left and right sideward flight were performed. The pilot was able to perform the maneuver rapidly, stabilize at 35 KTAS sideward flight, and maintain directional control throughout the maneuver. With the 33 degrees tail rotor rigging, greater than 10 percent control margins were available throughout the maneuver. This represents improved directional control from EDT-2. During EDT-2, lateral accelerations could not be conducted because of inadequate directional control.

40. Lateral reversal maneuvers were evaluated by stabilizing the aircraft in sideward flight at 35 KTAS and reversing direction to establish approximately 35 KTAS sideward flight in the opposite direction. The maneuver was performed at both a moderate rate and aggressively from left and right sideward flight. Figure 38 is a time history of moderate rate lateral reversal from 35 KTAS right sideward flight. The aircraft was controllable during the moderate rate reversal, altitude was maintained, and heading change throughout the maneuver was ± 15 degrees. Altitude could not be maintained during an aggressive lateral reversal and considerable pilot effort was required to maintain aircraft control (HQRS 8). When performing an aggressive lateral reversal in an NOE environment the pilot may be unable to maintain sufficient aircraft control to prevent settling into obstacles. Additionally, all lateral reversals were performed with the original tail rotor rigging of 33 degrees. The revised tail rotor rigging of 27 degrees would have provided only 0.1-inch left directional control margin during a moderate rate reversal and insufficient left directional control during an aggressive lateral reversal. The left directional control margin with a revised tail rotor rigging of 27 degrees will be insufficient for the performance of lateral reversal maneuvers. A reevaluation of the revised tail rotor rigging from 33 degrees to 27 degrees should be conducted. The operational requirement for lateral reversal maneuvers should be thoroughly investigated and clearly defined due to the hazardous nature of the maneuver.

Power Management

41. Power management characteristics of the YAH-64 were evaluated throughout EDT-4. The torque matching and turbine gas temperature (TGT) limiting features of the YT 700-GE-700R engines continued to be excellent.

42. The YAH-64 incorporated a pilot adjustable engine power control lever friction. During this evaluation the aircraft was operated with the adjustable friction full OFF. With the friction full OFF a force of approximately 7 pounds was required to operate both power levers. Precise engine power lever movement, such as during engine control unit (ECU) lockout operation, was difficult. The excessive control force required to operate the engine power control levers failed to meet the requirements of paragraph 10.3.3.2.3 of reference 7, appendix A by 2.0 pounds. The high inherent friction of the engine power control levers is a shortcoming previously reported.

43. The pilot engine control quadrant was located on the left console. Operation of the power levers, such as during ECU lockout operation, was made difficult because the pilot's arm contacted the armor protection on the pilot's seat when grasping the power levers in a normal fashion. The poor location of the pilot engine control quadrant is a shortcoming.

Mission Maneuvering Characteristics

44. The mission maneuvers evaluated during this test were selected primarily to evaluate the aircraft handling qualities during the terrain flight mode of operation. Pilot duties were limited to maneuvering of the aircraft only, and no mission scenario was simulated. The following standards for NOE and contour flight, as defined by reference 11, appendix A, were used during this evaluation.

NOE Flight: "Fly as close to the earth's surface as vegetation or obstacles permit."

Contour flight: "Maintain 20-foot obstacle clearance ± 10 feet, generally conforming to the contours of the earth."

45. Significant obstructions to the pilot's field of view were observed during NOE and contour flight. Obstructions included the CPG helmet which obstructed approximately 40 percent of the forward windscreen (with CPG seated at the design eye position), canopy reflections which were most evident when flying in haze or smoke, and the overhead circuit breaker panel which obstructed the pilot's field of view during a left banked turn. Distortion was also present in the canopy windows for approximately one inch on either side of each canopy frame structural member and further restricted the pilot's field of view. The restricted pilot's field of view caused by the above items was a shortcoming. The major obstruction to the pilot's field of view was the canopy frame structure shown by the shaded areas of figure 1 (previously reported in reference 1, appendix A as the top priority shortcoming). The structure blocked the pilot's field of view at the eleven and one-o'clock positions. During NOE flight, when attempting to fly between trees or other obstacles, the pilot was unable to accurately determine the height of the obstacles since they were obstructed from view at the time a height decision was required. As a result, the pilot flew 10 to 20 feet higher than the terrain would normally have allowed, thereby preventing NOE flight in accordance with reference 11. Similar restrictions to the pilot's field of view were observed during contour flight and the pilot could not maintain the aircraft within the specified 20 feet ± 10 feet obstacle clearance when making turns due to the poor field of view caused primarily by the canopy frame structure. The restricted pilot's field of view caused by canopy frame structure during NOE and contour flight (as defined by current aircrew training

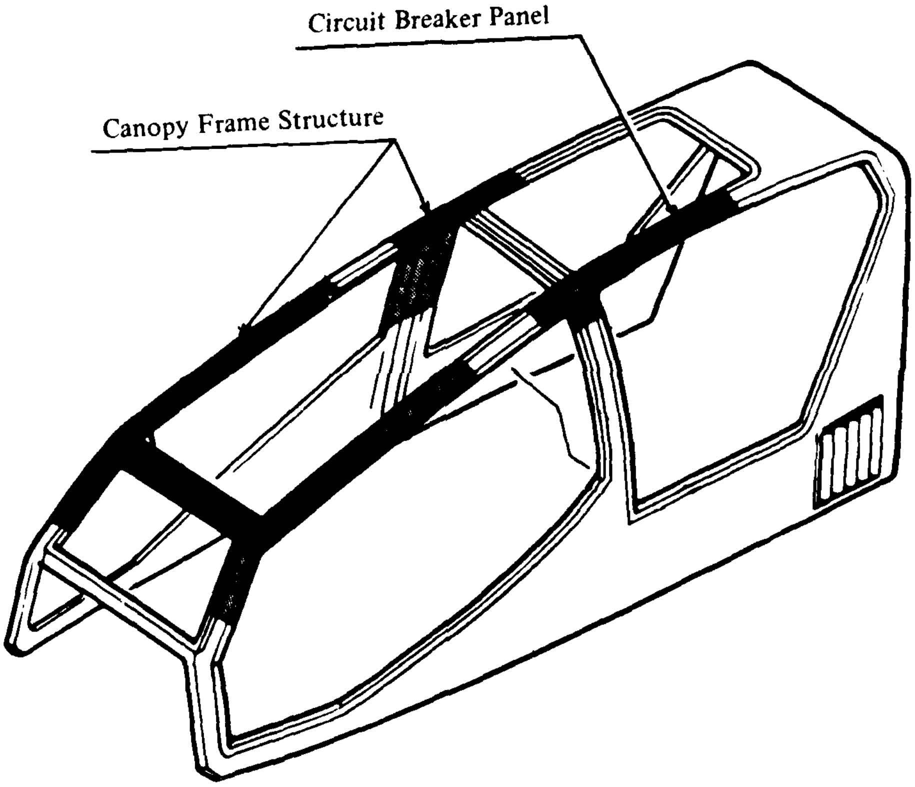

Figure 1. Canopy Frame Structure

manual) is a deficiency. The pilot's field of view during NOE and contour flight should be identified as a special area of interest for operational testing.

46. Masking and unmasking maneuvers were performed with and without the hover augmentation system (HAS) engaged. Pilot workload was significantly reduced during hover, masking and unmasking with the HAS engaged. A slight left lateral drift occasionally occurred after engagement of the HAS. Engagement of the HAS required the pilot to remove his hand from the collective control and reach forward on the left console to activate a toggle switch. The requirement for the pilot to remove his hand from the flight controls to engage the HAS is undesirable particularly when in the NOE environment. Additionally, an altitude-hold feature incorporated in the HAS would allow the pilot to briefly remove his hands from the controls to perform mission essential tasks (*i.e.*, change radio frequencies, adjust the PNVS, refer to maps, etc.), while in a masked position. The handling qualities of the YAH-64 during masking and unmasking manuevers are satisfactory for the attack helicopter mission. An altitude-hold feature should be considered for incorporation into the HAS. Provisions should also be incorporated to allow the pilot to engage the HAS without removing his hands from the flight controls.

Instrument Flight Capability

47. A limited IMC evaluation, which consisted of level flight, climbs, climbing and descending turns, and simulated radar vectoring, was conducted. Tests were performed under simulated IMC conditions (pilot wearing hood) with both DASE ON and OFF at airspeeds of 87 and 115 KCAS. The pilot was concerned only with aircraft control and did not attempt to navigate, perform approaches, change radio frequencies or make radio calls as would be required during actual IMC flight. The electronic attitude direction indicator (EADI) was inoperative and aircraft attitude information was obtained from the standby attitude indicator. The rate of turn indicator was also inoperative. The highest pilot work load occurred while trying to control airspeed with DASE OFF. Small, frequent longitudinal control inputs were required to maintain airspeed within ±10 KIAS (DASE OFF) particularly during climbing turns (HQRS 5-6). Power changes caused excitation of the three-axis oscillation and was most pronounced at 115 KCAS (see para 24). The pilot work load required in the longitudinal axis was considerably reduced with DASE ON (HQRS 3-4) with the major work load being for airspeed control. Factors contributing to the increased pilot work load in the longitudinal axis were previously discussed and are as follows: Excessive longitudinal control breakout force (plus friction) (para 12), poor trimability (para 15), neutral static longitudinal stability (para 20) and the divergent longitudinal long-period oscillation (para 26). A left roll attitude of 5 degrees was observed at 115 KCAS with zero sideslip as maintained by the DASE. This was uncomfortable to the pilot and may induce vertigo under actual IMC conditions. The uncomfortable and potentially vertigo-inducing left roll attitude at cruise airspeed during simulated IMC flight is a shortcoming. A thorough evaluation of the instrument flight capability of the YAH-64 should be conducted with production flight instruments installed.

Aircraft Systems Failures

Simulated Single Engine Failures:

48. Single engine failures were simulated by having the CPG retard one of the power levers to the idle position during various flight conditions as shown in table 1.

Collective pitch reduction was made after 2.0 seconds or at the activation of the low main rotor rpm warning. Representative time histories of simulated single engine failures during an IRP climb and at V_H is shown in figures 39 and 40, respectively. Aircraft response to the engine failure was mild and only small, instinctive control inputs were required to maintain aircraft attitude (HQRS 2). Available collective control delay time was only 1.2 seconds (0.8 seconds less than specification requirements); however, when the collective was reduced the main rotor speed was re-established at 100 percent in less than 3 seconds. The minimum main rotor speed attained was 90 percent. This was 3 percent below the minimum power on transient limit of 93 percent. Although aircraft response to the single engine failure was satisfactory to the pilot, single engine operation below 93 percent main rotor speed should be investigated to insure that structural limits will not be exceeded. The available collective control delay time failed to meet the 2.0 second requirement of paragraph 10.3.8.1.1 of reference 7, appendix A by 0.8 second at IRP.

49. The activation threshold of the low main rotor rpm warning, a previously reported deficiency, was evaluated during flight. The warning activated consistently at 95 percent main rotor speed (N_R) (as opposed to 91 percent previously reported in ref 3, app A) and is satisfactory for the attack helicopter mission.

50. The engine out warning system was evaluated by simulating the failure condition on the ground. The gas producer (N_G) warning activated at 63 percent and would provide an engine out warning in the event of a complete engine failure. The power turbine (N_P) warning activated consistently at 93 percent and provided a satisfactory engine out warning of a partial power engine malfunction. A previously reported deficiency (ref 3, app A) stated that adequate cues were not available to warn the pilot of a partial power engine malfunction which resulted in N_G stabilizing at idle speed (67-68 percent). The N_P warning activated at 71 percent N_G and will provide an engine out warning above idle N_G. The engine out warning system is satisfactory.

Digital Automatic Stabilization Equipment Failures:

51. Simulated single axis and total DASE failures were evaluated by turning off the respective axes or by total disengagements of the DASE. All controls were free for three seconds following the failure. Aircraft response to a single axis DASE failure was moderate. For a pitch axis failure, the aircraft response was a mild nose-down pitch. For a roll axis failure the aircraft response was a mild left roll. There was a negligible response to a yaw axis failure. The aircraft response to a total DASE failure was the three-axis short-term oscillation discussed in paragraph 24. The pilot required one-half inch lateral and longitudinal control inputs to maintain trim attitude with a total DASE failure. The requirement to use control inputs in excess of 0.25 inch to maintain the desired aircraft attitude failed to meet the requirements of paragraph 10.3.2.7.1 of the detailed specification. However, the aircraft response was controllable and is satisfactory.

Stabilator Failures:

52. Stabilator failures were evaluated at the conditions in table 1 in level flight by introducing a single stabilator actuator hardover. All controls were free for three seconds following the failure. Figure 41 is a typical time history of a stabilator failure. After the trailing edge of the stabilator had moved down approximately 10 degrees, the actuator position miscompare circuit automatically actuated and

stopped the stabilator travel. The aircraft response was a nose-down pitch of less than 10 degrees and the aircraft attitude was easily controlled after the three seconds. The minimum longitudinal control margin during recovery at 120 KCAS was in excess of 10 percent. Aircraft response following stabilator hardovers in level flight is satisfactory.

53. Takeoffs with the stabilator failed were performed. The stabilator was failed in the full trailing-edge-down position and both normal and level acceleration takeoffs were made. Figure 42 is a typical time history of a takeoff with a failed stabilator. The pilot was aware of the failure upon accelerating through 50 KCAS because of the abnormal nose-down pitch attitude of the aircraft. At 100 KCAS the pitch attitude was approximately 10 degrees nose down and approximately 40 percent of longitudinal control remained. The aircraft was easily controllable. The aircraft response to a takeoff and acceleration to 100 knots indicated airspeed (KIAS) with the stabilator failed in the full trailing-edge-down position (25 degrees) is satisfactory.

54. Landings were made with the stabilator failed in a high speed flight position (4 degrees trailing edge up). Figure 43 is a typical time history of a landing with the stabilator failed. The aircraft was easily controllable during both normal approaches and simulated autorotations with the stabilator failed. The pitch attitude during the approach was slightly more nose up than with the stabilator operating normally; however, it was considered satisfactory for a failure mode. The aircraft response to a landing with the stabilator failed in a high-speed position is satisfactory.

Generator Failures:

55. Generator failures were simulated in flight by turning off either the No. 1 or No. 2 generator switch. A No. 2 generator failure produced disengagement of the DASE. Disengagement of the DASE with failure of the No. 2 generator is a shortcoming. A No. 1 generator failure caused disengagement of the heading and attitude reference system (HARS), disengagement of the DASE, failure of the automatic mode of stabilator operation and activation of the low rpm/engine out audio tone. During NOE or instrument flight or when operating the PNVS such failures and erroneous warnings would adversely affect flight safety. The failure of these systems and erroneous activation of the engine out/low rpm audio tone with the failure of the No. 1 generator is a deficiency which should be corrected prior to operational testing. In addition, the planned electromagnetic interference (EMI) evaluation should be conducted prior to operational testing. The YAH-64 failed to meet the requirements of paragraph 10.3.2.7.8 of reference 7, appendix A in that failure of a single generator caused failure of the DASE.

VIBRATION CHARACTERISTICS

56. The vibration characteristics and canopy drumming of the YAH-64 were evaluated throughout the test program. Qualitative evaluations were made at both crew stations with vibration absorber removed and installed. Vibration characteristics are shown in figures 44 through 77, appendix E, and test conditions are shown in table 3.

57. The 4/rev (19.2 Hz) vertical vibration of 0.3 to 0.4g in rearward flight at airspeeds greater than 25 KTAS was objectionable to the pilot (VRS 6) (fig 44). However, at the same conditions the vibration at the copilot seat (fig 45) was not

Table 3. Vibration Test Conditions

Gross Weight (lb)	Longitudinal CG Location (fs)	Density Altitude (ft)	Flight Regime	Location[1]
14820	201.3 (Fwd)	440	Left and right sideward	Pilot seat and floor
14720	201.4 (Fwd)	280	Forward and rearward	Copilot seat and floor Aircraft cg
15640	202.2 (Fwd)	9720	Level	
14400	205.7 (Aft)	5840	Level[2]	
15600	202.3 (Fwd)	5700	Climb	Pilot and copilot seat Aircraft cg
15580	202.3 (Fwd)	4480	Descent	Pilot and copilot seat
14600	202.0 (Fwd)	1360	Level	
14860	202.0 (Fwd)	5420	Level	
15740	205.2 (Aft)	5020	Level	
15740	202.0 (Fwd)	6380	Level	

[1] Three-axis vibration at each location
[2] Vibration absorber out

objectionable. In right sideward flight at airspeeds greater than 15 KTAS the lateral vibration of 0.2 to 0.3g (fig 49) was objectionable (VRS 4). Similarly, the vibration at the copilot's station under the same flight conditions was not objectionable (fig 50).

58. The vibration in climb and descent were evaluated at both the pilot and CPG stations (figs 54 through 59) and was found to be satisfactory.

59. The vibration characteristics were evaluated in level flight at the pilot station (figs 60 through 72). As in low-speed flight the vibration levels at the CPG station were generally lower than at the pilot station. The 4/rev lateral vibration of greater than 0.2g at the pilot's seat in level flight at airspeeds less than 50 KCAS (figs 64 and 66) were objectionable (VRS 4). Also the 4/rev vertical vibration of 0.25g at the pilot's seat in level flight at airspeeds greater than 117 KCAS was objectionable (VRS 5).

60. The vibration characteristics during the termination of the approach were qualitatively evaluated. These vibrations were similar to those encountered during rearward flight at airspeeds greater than 25 KTAS and were objectionable to the pilot (VRS 6).

61. The vibration characteristics were evaluated with the vertical vibration absorber removed. Figures 73 through 77 depict the vibration levels during level flight with the absorber removed. Although the 4/rev vertical vibration levels are slightly greater at high speed for similar gross weight and density altitude, there were no qualitative differences noticed at the pilot or CPG station. During climbing, descending, and low-speed flight there were no significant differences noticed with the vibration absorber removed.

62. The vibration characteristics and canopy drumming of the YAH-64 are generally improved from EDT-2. While canopy drumming was not objectionable during this test, the vibration characteristics remain objectionable during several flight conditions. The objectionable 4/rev vertical vibration in rearward flight at airspeeds greater than 25 KTAS, lateral vibration in right sideward flight at airspeeds less than 15 KTAS, lateral vibration in level flight at airspeeds less than 50 KCAS and greater than 117 KCAS, and the vertical vibration during the termination of the approach remain a shortcoming. Efforts to reduce the objectionable 4/rev vibration should continue.

HUMAN FACTORS

Cockpit Evaluation

63. The pilot cockpit of the YAH-64 was evaluated throughout the test program. This included an evaluation of cockpit design with primary emphasis on instrument displays, switch function and design, and ease of operation of system controls. Pilot comfort was also considered. Two previously reported shortcomings, the need for additional cockpit storage area and provisions for passing mission essential items between cockpits, should be further evaluated during operational testing.

64. The engine torque indicator vertical scale was limited to 120 percent. The airworthiness release for EDT-4 allowed a transient torque limit of 125 percent. The

engine torque indicator should have an adequate range to span the allowable aircraft limits. The lack of adequate display capability of the engine torque indicator vertical scale is a shortcoming.

65. The collective pitch control friction was difficult to adjust to the desired level. A small adjustment of the friction twist grip frequently produced too much friction. The friction level was not constant throughout the range of collective control travel. The design incorporates a friction-bar slider assembly which is attached to the cockpit floor and to the collective lever. The friction bar slider assembly was susceptible to damage during routine maintenance and often was found to be slightly bent which produced a ratcheting effect when applying collective pitch. Maintenance action had been taken several times during the test, however, the problem was not corrected. The poor design of the collective pitch control friction mechanism is a shortcoming.

66. During normal operation, the Marconi instruments displayed only three segment lights at any one time and did not show all green segments to indicate the limits of the normal range of operation. This is different from Marconi instruments in other Army aircraft and detracted from the pilot's ability to rapidly cross check instrument indications. The failure of the Marconi instruments to display the full green range during normal operation is a shortcoming.

67. The rocket panel displays, the Marconi instrument indications, and the caution, warning and advisory panel segment lights were unreadable in direct sunlight. In order to determine the indications being displayed the pilot had to remove his hand from the flight controls and shade the appropriate instrument or panel. This was annoying to the pilot and undesirable particularly in a high work load environment. The washout of the rocket panel displays, Marconi instrument indication, and caution, warning and advisory panel segment lights in direct sunlight is a shortcoming previously reported.

68. The environmental control unit (ENCU) was evaluated during flight and was found to be much more effective at the CPG station. When the pilot adjusted the control for a comfortable temperature, the CPG station was either too hot or too cold. When attempting to heat the cockpit, the CPG station was too warm; when cooling the cockpit, the CPG station was too cool. Tests were not conducted at extreme temperatures. Operation of the ENCU should be a special area of interest during climatic testing. The inability to maintain both crew stations at the same temperature, using the ENCU, is a shortcoming.

69. The lower green segment light of each Marconi instrument vertical scale remained illuminated during operation. This green light indicated that electrical power was being supplied to the instrument. In addition to the pilot initiated test feature, the illumination of any light on the vertical scale would show that power was being supplied. The illumination of the lower segment light was confusing to the pilot and detracted from his cross check of instrument indications. The constant illumination of the lower green segment light on the Marconi vertical scale is a shortcoming previously reported.

70. During auxilary power unit (APU) start the APU ON advisory light illuminates prior to reaching 100 percent N_G. This is a false indication of APU status and could lead to automatic shutdown of APU if systems (*e.g.*, generator) are activated

prematurely. The illumination of the APU ON advisory light prior to the APU stabilizing at 100 percent rpm is a shortcoming previously reported.

71. The pilot's cyclic grip incorporated eight different functions activated by thumb switches. In addition, some additional functions are required to be activated simultaneously such as the weapons action switch and the trigger switch. An extreme reach was also required for operation of the trim release switch. A pilot with a smaller than normal hand will have difficulty in operating the switches on the cyclic and will frequently be required to reposition his hand on the cyclic grip in order to perform the necessary functions. The poor anthropometric design of the pilot's cyclic grip is a shortcoming previously reported.

72. Various annoying tones were present in the intercom system throughout the test. Flying with the PRI LTS circuit breaker out helped to alleviate some of the tones; however, they still were not reduced to a satisfactory level. The tone was continuous and appeared to be of an EMI nature. The annoying tone present in the intercom system is a shortcoming. The planned EMI evaluation should be conducted prior to operational testing.

73. The pilot seat currently installed in the YAH-64 is adjustable only in the vertical axis. The directional pedals are also adjustable, however, the cyclic and collective controls are not. During this test a 70 percentile height aviator was unable to obtain a comfortable seating position with reference to the cyclic and collective controls. It wll be extremely difficult for a person near the extreme range of US Army anthropometric measurements to achieve a comfortable seating position. The difficulty in attaining a comfortable seating position with reference to the cyclic and collective controls is a shortcoming. Consideration should be given to installing a longitudinal adjustment for the pilot seat.

74. The operation of the HARS control switch was difficult. When positioning the switch to or from the OPER position it was necessary to pull out on the switch and simultaneously turn it to the desired position. Occasionally the pilot fingers would slip off the switch and he was required to remove his glove to operate the switch properly. The difficulty in operating the HARS control switch is a shortcoming.

75. Frequent, intermittent illumination of the master caution and engine fuel pressure warning lights was observed during engine start. Though no related problems were encountered during engine start the indications were distracting to the pilot. During each engine start the pilot was required to reset the master caution light 6 to 8 times due to the intermittent illumination of the engine fuel pressure warning lights. The intermittent illumination of the master caution and engine fuel pressure warning lights during engine start is a shortcoming.

76. Illumination of a green advisory light on the caution, warning and advisory panel caused the master caution light to illuminate. The illumination of the amber master caution light is actually an erroneous indication, in this instance, since it alerts the pilot to a malfunction when actually none has occurred. The illumination of the master caution light with green advisory segment lights is a shortcoming.

77. The anti-ice panel was located on the aft portion of the pilot's left console. Several of the switches were obstructed from view by the aircraft structure and the armor plating on the left of the pilot's seat. The switches were difficult to reach and some of the switch labeling was hidden from view. When operating the anti-ice

panel switches the pilot will be required to rely on feel only (for several switches) and, due to the awkward location of the panel, may inadvertently activate other switches located on or near the anti-ice panel. The poor location of the anti-ice panel switches is a shortcoming.

78. During engine start the Marconi torque indicator displayed a full scale deflection (120 percent). This is an undesirable and erroneous indication and is distracting to the pilot. The full scale illumination of the Marconi engine torque indicator during engine start is a shortcoming previously reported.

79. The lack of adequate cockpit storage area and the lack of provisions to allow the passing of mission essential equipment between cockpits were two previously reported shortcomings. Though not thoroughly evaluated during EDT-4 it appeared that the cockpit storage area would not be readily accessible in flight and that the small slot on the lower left portion of the blast shield would be inadequate for passing items between cockpits. The requirement for additional cockpit storage area and provisions for passing mission essential items between cockpits should be further evaluated during operational testing.

Fuel Management

80. The fuel management system was evaluated primarily to determine if the previously reported deficiency (ref 3, app A) had been corrected. Due to the inoperative fuel transfer pump, the design changes incorporated to correct the previous deficiency, could not be evaluated. Low fuel transfer rate, a previously reported shortcoming, also could not be evaluated. A fuel transfer capability sufficient to supply adequate fuel to both engines operating at IRP should be demonstrated with the fuel transfer pump installed in the aircraft. A complete description of the YAH-64 fuel system is included in appendix B.

81. The design of the pilot's fuel control panel (fig 2) was evaluated throughout the test program. The switch labeling and function (although redesigned since EDT-2) remain confusing and the engine fuel switches were difficult to see with the engine power control levers in the full aft position. Normal operation of the fuel system required frequent pilot attention to the fuel control panel in order to maintain comparable fuel levels in each tank. When attempting to transfer fuel or use fuel from one tank only, no immediate indication was available to show the direction of transfer or the failure of the transfer pump. The TK SEL switch does not always indicate the correct operation of the system. With the boost pump ON (required for operation above 10,000 feet, pressure altitude) the TK SEL switch is disabled and may be positioned to the FROM FWD position when in fact, fuel will be supplied to both engines from the aft tank. Additionally, the TK SEL switch can be inadvertently activated by the pilots sleeve or glove. Ideally, operation of the fuel management system should be such that the pilot would not be required to activate various switches on the fuel control panel during flight unless a malfunction or emergency condition existed. The poor design of the pilot's fuel control panel is a previously reported shortcoming.

RELIABILITY AND MAINTAINABILITY

82. The reliability and maintainability features of the YAH-64 aircraft were evaluated throughout the test. Nineteen Equipment Performance Reports (EPRs)

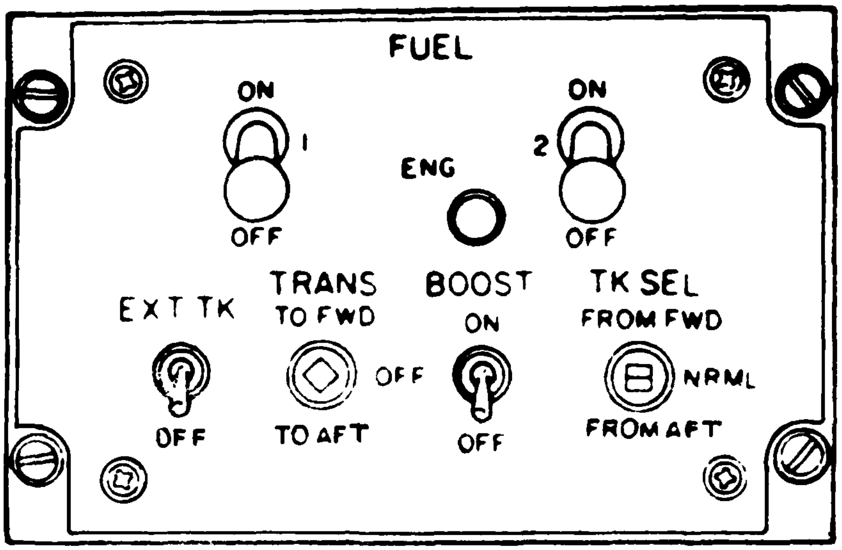

Figure 2. Pilot's Fuel Control Panel

were prepared and submitted during this test and are listed in appendix G. This section is only intended to document those undesirable reliability and maintainability features encountered.

a. The master caution light randomly failed to illuminate with the illumination of a caution or warning segment light. The random failure occurred a minimum of eight times during six flights and occurred in conjunction with the illumination of the following caution panel segment lights: SAS, F FEEL, STAB FAIL, GEN 1, GEN 2. The problem appeared initially at the pilot station but occurred once at the CPG station also. Maintenance action was taken, to include replacement of the pilot's warning panel, but subsequent flights showed that the problem had not been corrected. Failure of the master caution light to illuminate, primarily when the pilot's attention is directed outside the aircraft (as during NOE flight) will be hazardous since the pilot may not be alerted to a potentially critical malfunction. Random failure of the master caution light to illuminate with the illumination of some caution or warning panel segment light is a deficiency.

b. The fuel transfer pump failed to operate during the test program. Frequent maintenance action, including replacement of the fuel transfer pump, was performed. All attempts to correct the problem were unsuccessful. All tests were flown without a fuel transfer capability and fuel management was accomplished by using the tank select function. The inoperative fuel transfer pump is a shortcoming which should be corrected prior to operational testing.

c. The heading and attitude reference system (HARS) failed to align accurately during the test. Alignments, both normal and fast, were inconsistent and indicated heading varied as much as ±20 degrees from actual runway heading. The inconsistent and inaccurate alignment of the HARS is a shortcoming which should be corrected prior to operational testing.

d. The Marconi scale for the No. 2 engine power turbine speed (N_P) showed 102 percent (caution range) throughout the test. The main rotor speed and No. 1 engine N_P both read 100 percent. The illumination of the caution range is distracting to the pilot since it indicates an overspeed condition. The indication of the No. 2 engine N_P at 102 percent with the main rotor speed at 100 percent is a shortcoming.

e. The rotor brake failed to hold with both engines at idle. During the start of the second engine, the rotor brake began to slip as 60 percent N_G was reached. To initially set the rotor brake, the switch had to be moved to the BRAKE position then rapidly through OFF to the LOCK position. The requirement to move the switch through the OFF position may contribute to some loss of brake pressure. The failure of the rotor brake to hold with both engines at idle is a shortcoming previously reported. The performance of the rotor brake failed to meet the requirements of paragraph 3.7.1.1.7 of reference 7, appendix A, in that the rotor brake failed to hold with both engines at idle.

f. An excessive accumulation of oil was noted on the main transmission deck and in the upper fairing maintenance access area. This accumulation was due primarily to the moderate leakage of the engine nose gearboxes (EPR 80-03-7, app G). A leak in the main transmission left input seal, at the rate of 60 drops per minute, was noted prior to engine start with the auxiliary power unit in

operation (EPR 80-03-15, app G). The excessive accumulation of oil on the main transmission deck and in the upper fairing maintenance access area is a shortcoming.

g. The correct fluid service level of the primary and utility hydraulic reservoirs was difficult to determine. It was impossible to see any decals or markings on the primary reservoir. The utility reservoir was marked with a decal indicating normal range but no minimum or maximum service level was indicated. Additionally, the decal marking was subject to deterioration and appeared to be peeling off. The difficulty in determining the correct service level of the primary and utility hydraulic reservoirs is a shortcoming previously reported.

h. Hydraulic fluid samples were being taken from the return side of the primary and utility ground service panel. The fluid taken from this point is not representative since it does not circulate throughout the system. No means were available to take an accurate sample of primary or utility hydraulic fluid. The lack of an acceptable method for sampling the primary and utility hydraulic fluid is a shortcoming.

i. The primary and utility hydraulic reservoirs were frequently found to be overserviced during the preflight inspection. It appeared that the ground hydraulic service equipment overserviced the reservoirs when operating the aircraft with ground hydraulic power only. There was no method for bleeding the reservoir down to the proper service level prior to flight. During APU start some of the hydraulic fluid would be automatically dumped overboard through a drain line; however, it was not sufficient to return the reservoir to the proper service level. The lack of a method for bleeding an overserviced primary or utility hydraulic reservoir is a shortcoming.

j. A temporary loss of electrical power was observed when transitioning from aircraft power to external power. Various system failure indications were noted, to include automatic stabilator and engine out warnings, due to the power transient. Electrical power was restored to the aircraft by activating the EXT PWR RESET switch. The temporary loss of electrical power when transitioning from aircraft power to external power is a shortcoming previously reported.

k. During preflight it was extremely difficult to determine the engine oil levels without first opening the engine cowlings. A small piece of polished metal had been installed on the inside of the cowling which allowed the pilot to look up through the fire access door to see the sight gauge. Due to oil accumulation, scratches on the polished metal mirror, wiring harnesses, and insufficient lighting, it was extremely difficult to accurately determine the engine oil level using this method. The difficulty in accurately determining the engine oil levels without opening the engine cowlings is a shortcoming previously reported.

l. The fire bottle discharge lights, located on the fire test panel, failed to illuminate with activation of the PRESS TO TEST switch. The lights are designed to illuminate to advise the pilot when a fire bottle has been discharged. The PRESS TO TEST function is a preflight check to check the operation of the lights only. The failure of the fire bottle discharge lights to illuminate with the activation of the PRESS TO TEST switch is a shortcoming.

m. Activation of the rotor brake switch during engine shutdown caused activation of the automatic stabilator failure audio tone. This occurrence was

observed on the last three flights during EDT-4 and is an indication of EMI. The initiation of the automatic stabilator audio tone with the activation of the rotor brake switch during shutdown is a shortcoming. The planned EMI evaluation should be conducted prior to operational testing.

AIRCRAFT PITOT-STATIC SYSTEM

83. Calibration of the ship's airspeed system was accomplished during level, climbing, and descending flights. Two calibrated pace aircraft were used as speed references. Data from these calibrations are presented in figures 78 through 80, appendix E.

84. The YAH-64 has two ship's airspeed systems. One utilizes the left-hand pitot tube and the other uses the right-hand one. Both systems use the same two static ports located on either side of the fuselage. The airspeed systems functioned satisfactorily.

CONCLUSIONS

GENERAL

85. Based on the EDT-4 flight test of the YAH-64 helicopter, the following conclusions were reached:

a. Major improvements in aircraft handling qualities as well as some improvement in performance and vibration characteristics have been made since EDT-2.

b. The new empennage and tail rotor configuration has made a significant improvement in the pitch attitude of the aircraft during approach and IRP climbs, (para 16).

c. Low-speed flight characteristics including directional control margins have been greatly improved (para 35, 36, 37).

d. Short-period dynamic stability characteristics have been significantly improved, particularly in high speed flight (para 24).

e. Directional control margins with the revised directional control stop may be inadequate at high gross weight and high density altitudes and during lateral reversal maneuvers (para 35 and 40).

f. The removal of the vertical vibration absorber did not significantly affect the vibration levels (para 61).

g. Nineteen Equipment Performance Reports have been submitted during this test (para 83).

h. Three deficiencies have been identified.

i. Forty-four shortcomings have been identified.

DEFICIENCIES

86. The deficiencies reported herein are not necessarily intended to bar type classification per AR 310-25 (see app D for definition of deficiency used in this report). The following deficiencies (in order of importance) were identified:

a. The disengagement of the HARS, DASE, and automatic operation of the stabilator, and erroneous activation of the engine out/low rotor speed audio tone with failure of the No. 1 generator (para 55).

b. Random failure of the master caution light to illuminate with the illumination of some caution or warning panel segment lights (para 82a).

*c. The restricted pilot's field of view caused by canopy frame structure during NOE and contour flight (para 45).

*Previously reported as a shortcoming.

SHORTCOMINGS

87. The most significant shortcomings found during this test are listed in order of relative importance (see app D for definition of shortcoming).

a. The inoperative fuel transfer pump (para 82b).

*b. The poor design of the pilot's fuel control panel (para 81).

c. The inconsistent and inaccurate alignment of the HARS (para 82c).

d. The absence of SAS pitch rate damping at load factors greater than 1.6 due to saturation of the pitch SAS actuator (para 22).

**e. The restriction to the pilot's field of view caused by window edge distortion, the overhead circuit breaker panel, canopy reflections, CPG helmet, and the PNVS turret (para 32 and 45).

**f. The objectionable 4/rev vertical vibration in rearward flight at airspeeds greater than 25 KTAS, lateral vibration in right sideward flight at airspeeds greater than 15 KTAS, lateral vibration in level fight at airspeeds less than 50 KCAS and greater than 117 KCAS and the vertical vibration during termination of the approach (para 62).

*g. Disengagement of the DASE with failure of the No. 2 generator (specification noncompliance) (para 55).

*h. The poor trimability at airspeeds between 85 and 110 KCAS (para 15).

i. The uncommanded aircraft response when retrimming with the attitude hold engaged (para 19).

j. The uncomfortable and potentially vertigo-inducing left roll attitude at cruise airspeed during simulated IMC flight (para 47).

*k. The excessive longitudinal breakout force (specification noncompliance) (para 12).

*l. The abrupt longitudinal control trim shift from 12 to 18 KTAS (para 35).

*m. The large, abrupt lateral control trim shifts between 16 and 24 KTAS rearward and between 13 and 18 KTAS forward flight (para 34).

*n. The inadvertent directional control inputs during brake application (para 28).

o. The lack of adequate display capability of the engine torque indicator vertical scale (para 64).

*Previously reported as a shortcoming.
**Previously reported as a deficiency.

p. The poor design of the collective pitch control friction mechanism (para 65).

q. The inability to maintain both crew stations at the same temperature using the ENCU (para 68).

*r. The lack of a reliable indication of parking brake status (para 30).

s. The indication of the No. 2 engine N_p at 102 percent with the main rotor speed at 100 percent (para 82d).

t. The improper operation of the automatic stabilator during minimum power descents at 50 and 60 KCAS (para 17).

u. The lack of an acceptable method for sampling the primary and utility hydraulic fluid (para 82h).

*v. The high inherent friction of the engine power control levers (specification noncompliance) (para 42).

w. The annoying tone present in the intercom system (para 72).

88. Additional shortcomings found are listed below:

*a. The failure of the rotor brake to hold with both engines at idle (specification noncompliance) (para 82e).

b. The failure of the Marconi instruments to display the full green range during normal operation (para 66).

c. The excessive accumulation of oil on the main transmission deck and in the upper fairing maintenance access area (para 82f).

*d. The difficulty in determining the correct service level of the primary and utility hydraulic reservoirs (para 82g).

e. The lack of a method for bleeding an overserviced primary or utility hydraulic reservoir (para 82i).

*f. The poor location of the pilot engine control quadrant (para 43).

*g. The temporary loss of electrical power when transitioning from aircraft power to external power (para 82j).

*h. The washout of the rocket panel displays, Marconi instrument indications and caution, warning and advisory panel segment lights in direct sunlight (para 67).

*i. The poor location of the tail wheel unlock light (para 29).

*j. The constant illumination of the lower green segment light on the Marconi vertical scale (para 69).

*Previously reported as a shortcoming.

*k. The illumination of the APU ON advisory light prior to the APU stabilizing at 100 percent rpm (para 70).

*l. The poor anthropometric design of the pilot's cyclic grip (para 71).

*m. The difficulty in accurately determining the engine oil levels without opening the engine cowlings (para 82k).

*n. The difficulty in attaining a comfortable seating position with reference to the cyclic and collective controls (para 73).

o. The difficulty in operating the HARS control switch (para 74).

p. The intermittent illumination of the master caution and engine fuel pressure warning lights during engine start (para 75).

q. The illumination of the master caution light with green advisory segment lights (para 76).

r. The poor location of the anti-ice panel switches (para 77).

*s. The full scale illumination of the Marconi engine torque indicator during engine start (para 78).

t. The initiation of the automatic stabilator audio tone with the activation of the rotor brake switch during shutdown (para 82m).

u. The failure of the fire bottle discharge lights to illuminate with the activation of the PRESS TO TEST switch (para 82l).

SPECIFICATION COMPLIANCE

89. The YAH-64 was found not to be in compliance with the following paragraphs of the Phase 2 Advanced Attack Helicopter System Specification AMC-SS-AAH-H10000A. Additional specification noncompliance, beyond the scope of this evaluation, may exist.

a. 3.7.1.1.7 Failure of the rotor brake to hold with both engines at idle (para 82e).

b. 10.3.2.1.1 Excessive longitudinal control breakout force (plus friction) (para 12).

***c. 10.3.2.2 Excessive lateral control force gradient (para 13).

***d. 10.3.2.5 Longitudinal control force could not be trimmed to zero with the CPG cyclic control stick extended (para 12).

*Previously reported as a shortcoming.
***Considered acceptable.

***e. 10.3.2.7.1 Control inputs in excess of 0.25 inch were required to maintain the desired aircraft attitude with disengagement of the DASE (para 51).

f. 10.3.2.7.8 Failure of a single generator caused disengagement of the DASE (para 55).

g. 10.3.3.2.3 Excessive control force required to operate the engine power control levers (para 42).

***h. 10.3.8.1.1 Insufficient available collective delay time following single engine when operating at IRP (para 48).

***Considered acceptable.

RECOMMENDATIONS

90. The following recommendations are made:

a. Correct the deficiencies listed in paragraphs 86a and 86b prior to conducting operational testing.

b. Identify field of view as a special area of interest for operational testing (para 32 and 45).

c. Continue efforts to reduce the objectionable 4/rev vibrations (para 62).

d. Correct the shortcomings listed in paragraphs 87a, 87c, and 87d prior to operational testing.

e. Demonstrate a fuel transfer capability, sufficient to supply adequate fuel to both engines operating at IRP, with the fuel transfer pump installed in the aircraft (para 80).

f. Conduct the planned electromagnetic interference evaluation prior to operational testing (para 55, 72, and 82m).

g. Evaluate the need for additional cockpit storage area and provisions for passing mission essential items between cockpits during operational testing (para 79).

h. Investigate the structural implications of single engine operation below 93 percent main rotor speed (para 48).

i. Correct the remaining shortcomings listed in paragraphs 87 and 88 prior to production.

j. Reevaluate the decision to revise tail rotor rigging from 33 degrees to 27 degrees left directional control (para 35 and 40).

k. Conduct further evaluation of the instrument flight capability of the YAH-64 configured with production flight instruments (para 47).

l. Investigate and clearly define the operational requirement for lateral reversal maneuvers (para 40).

m. Reevaluate the yaw oscillations at airspeeds between 40 and 70 KCAS with the ADS providing sideslip information to the DASE (para 18).

n. Identify ENCU operation as a special area of interest during climatic testing (para 68).

o. Consider incorporating an altitude-hold feature in the hover augmentation system and the attitude hold mode of the DASE (para 19 and 46).

p. Incorporate provisions to allow the pilot to engage the hover augmentation system without removing his hands from the flight controls (para 46).

q. Consider installing a longitudinal seat adjustment for the pilot seat (para 73).

APPENDIX A. REFERENCES

1. Final Report, USAAEFA Project No. 74-07-2, *Development Test 1, Advanced Attack Helicopter Competitive Evaluation, Hughes YAH-64 Helicopter*, December 1976.

2. Final Report, USAAEFA Project No. 77-36, *Engineer Design Test 1, Hughes YAH-64 Advanced Attack Helicopter*, September 1978.

3. Final Report, USAAEFA Project No. 78-23, *Engineer Design Test 2, Hughes YAH-64, Advanced Attack Helicopter*, June 1979 (undistributed, awaiting AVRADCOM position letter).

4. Letter, AVRADCOM, DRDAV-DI, 1 August 1980, subject: Engineer Design Test 4 (EDT-4) of YAH-64 Advanced Attack Helicopter (AAH).

5. Test Plan, USAAEFA Project No. 80-03, *Engineer Design Test 4, YAH-64 Advanced Attack Helicopter*, September 1980.

6. Letter, AVRADCOM, DRDAV-D, 28 October 1980, revised 5 November 1980, subject: Airworthiness Release for the Engineer Design Test (EDT-4) of the YAH-64, S/N 77-23258.

7. YAH-64 Phase 2 Advanced Attack Helicopter Systems Specification, Hughes Helicopters, AMC-SS-AAH-H100000A, 10 December 1976.

8. Draft Contractor's Flight Manual, Report No. 77-TM-8001-2, October 1979.

9. Flight Test Manual, Naval Air Test Center, FTM No. 102, *Helicopter Performance*, 28 June 1968.

10. Flight Test Manual, Naval Air Test Center, FTM No. 101, *Helicopter Stability and Control*, 10 June 1968.

11. Aircrew Training Manual, Department of the Army, No. TC 1-136, *Attack Helicopter*, October 1978, with change 3, 3 August 1979.

APPENDIX B. DESCRIPTION

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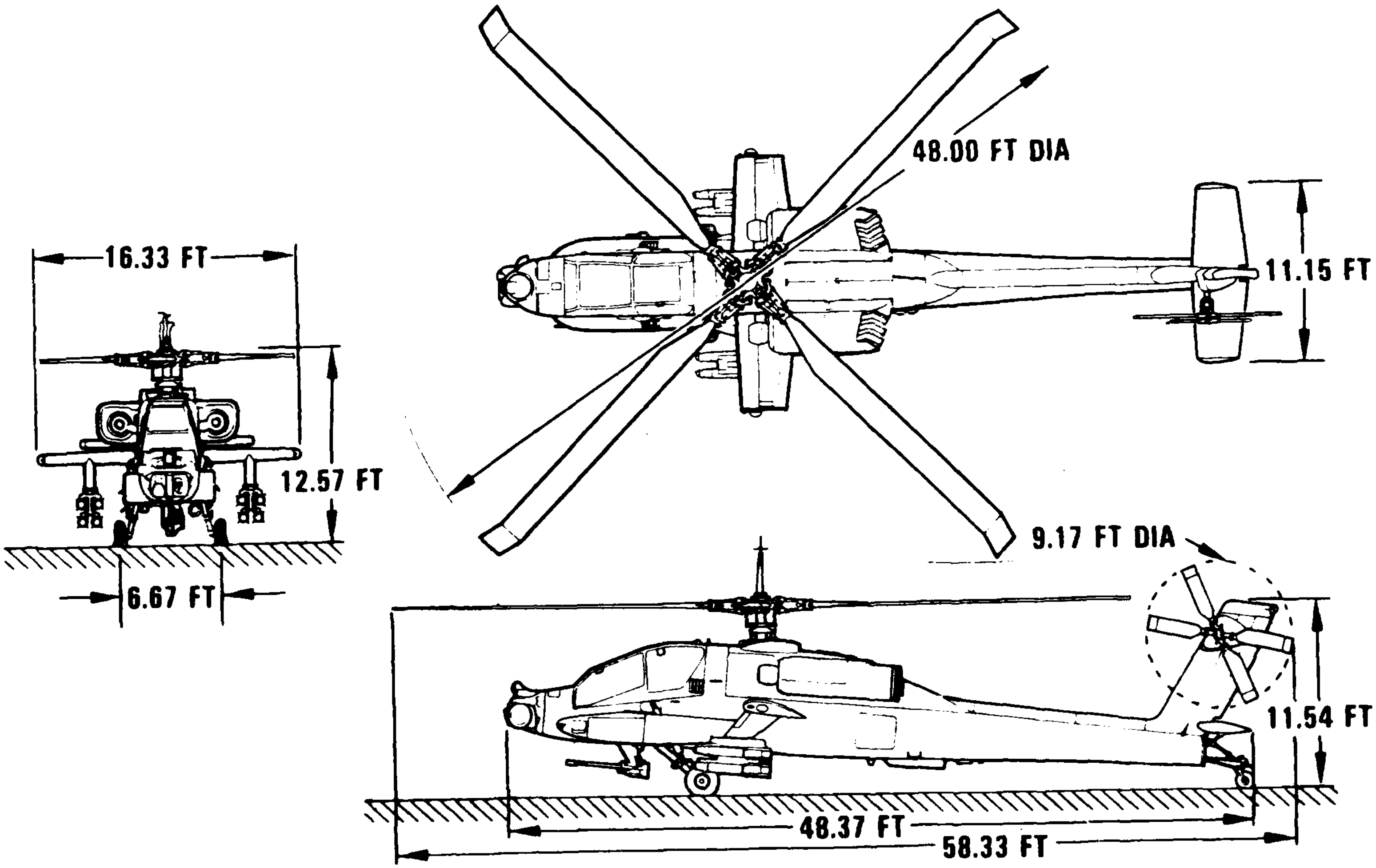


Figure 1. Aircraft Dimensions

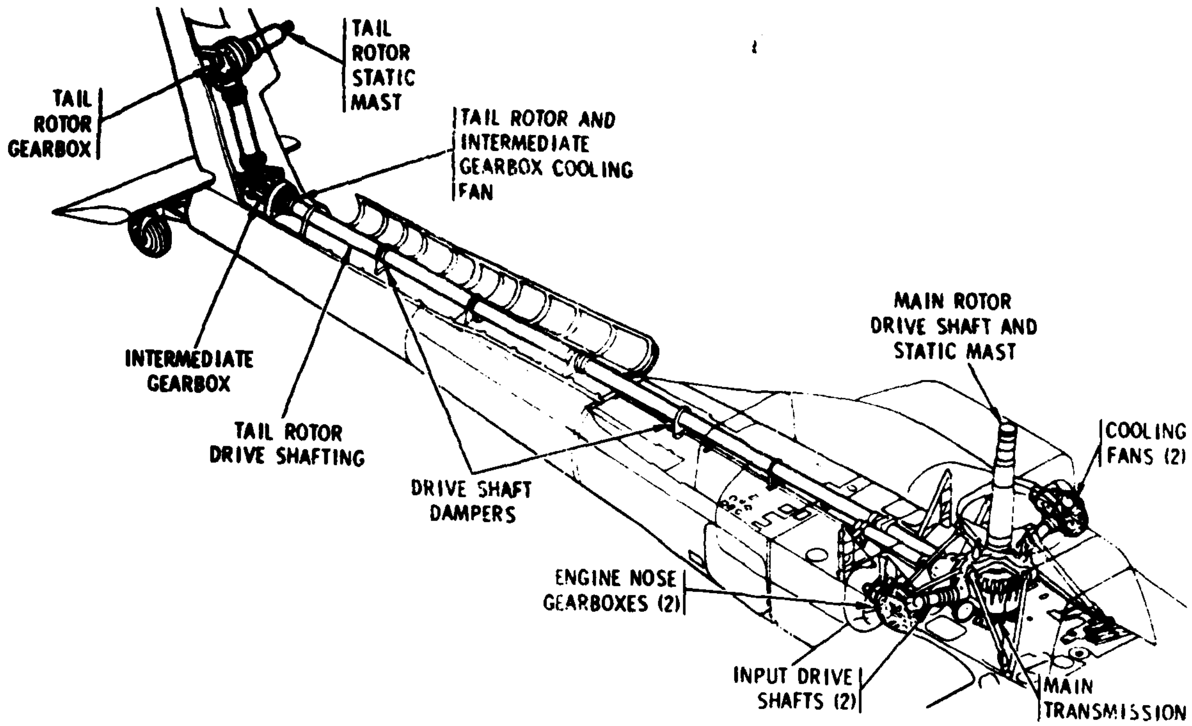


Figure 2. Powertrain

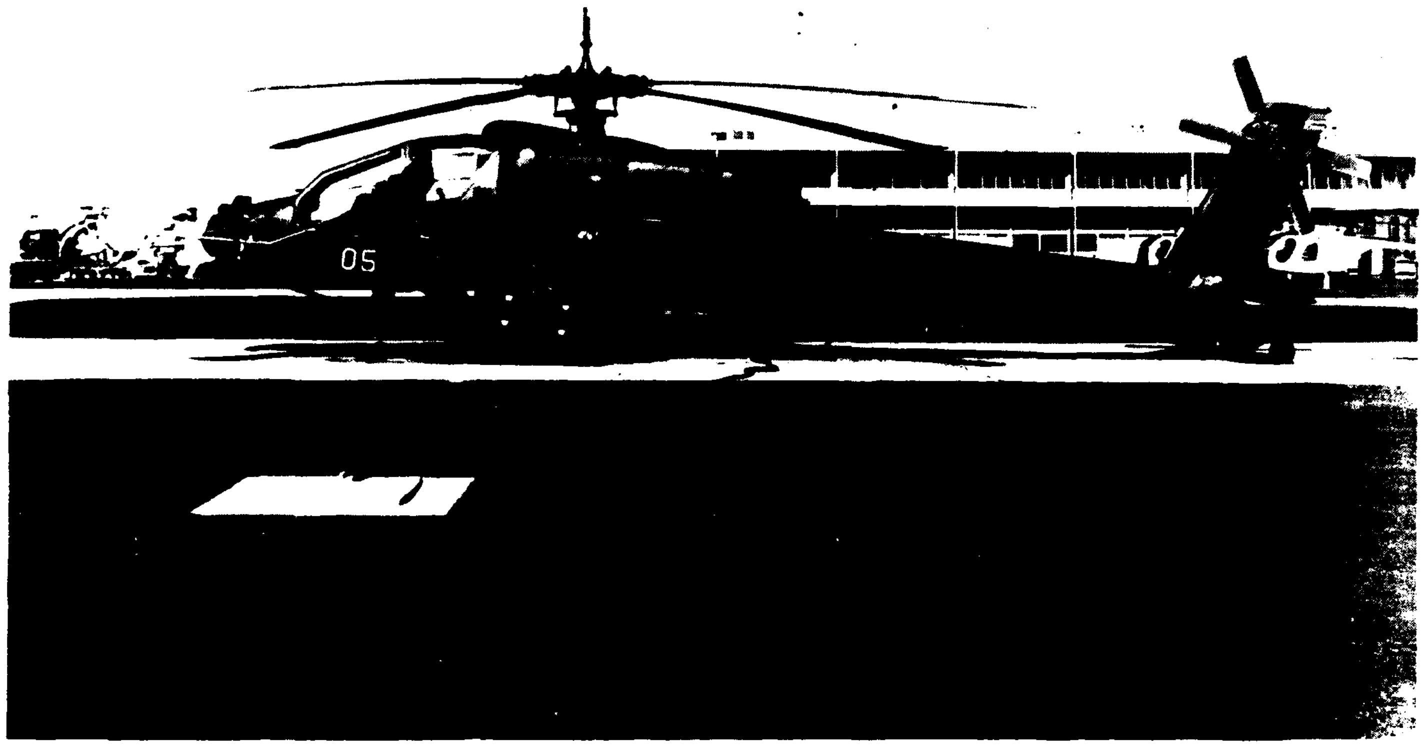


Photo 1. Left-Side View

Photo [illegible] Left Front Quartering View

Photo 3 Left Rear Quartering View

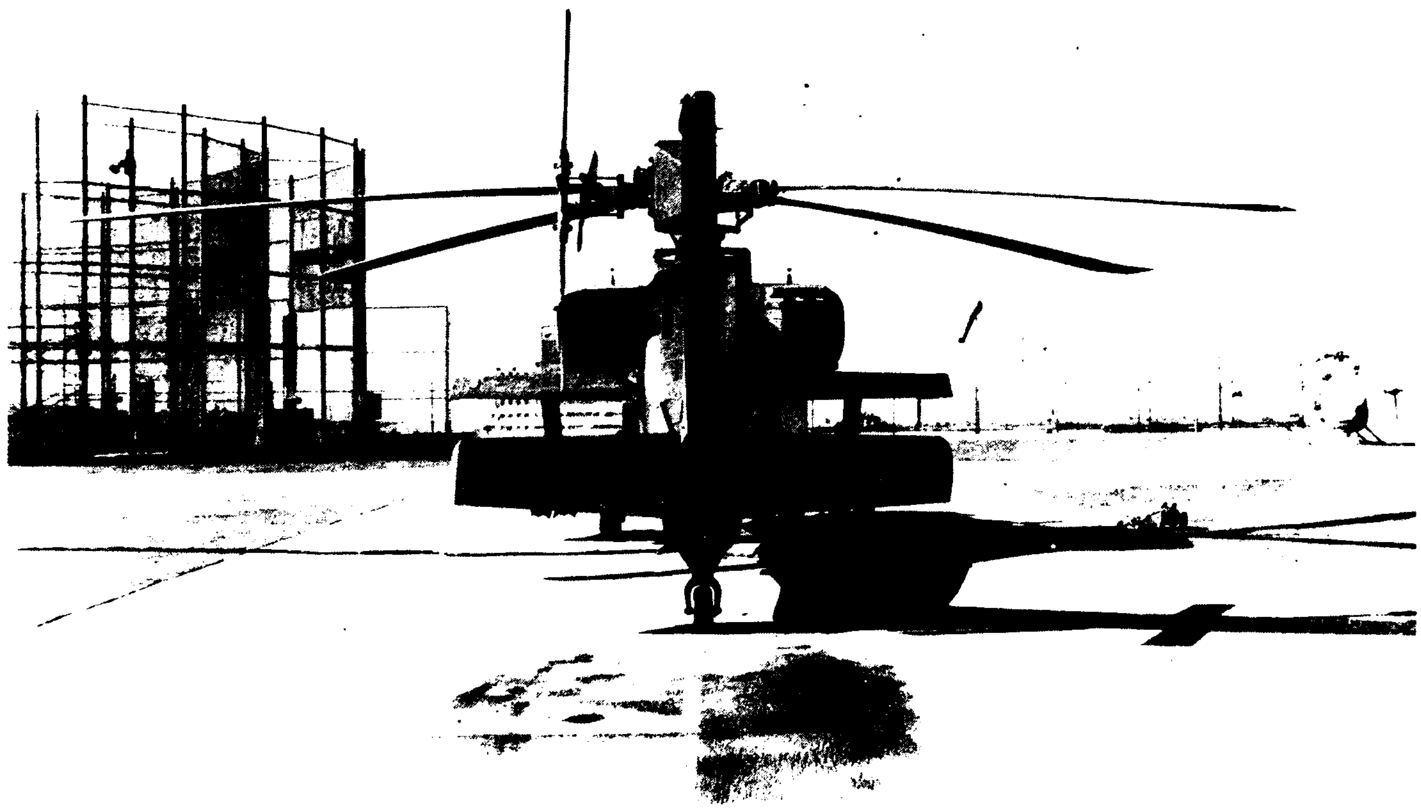


Photo 4. Rear View

Photo 5. Right-Front Quartering View

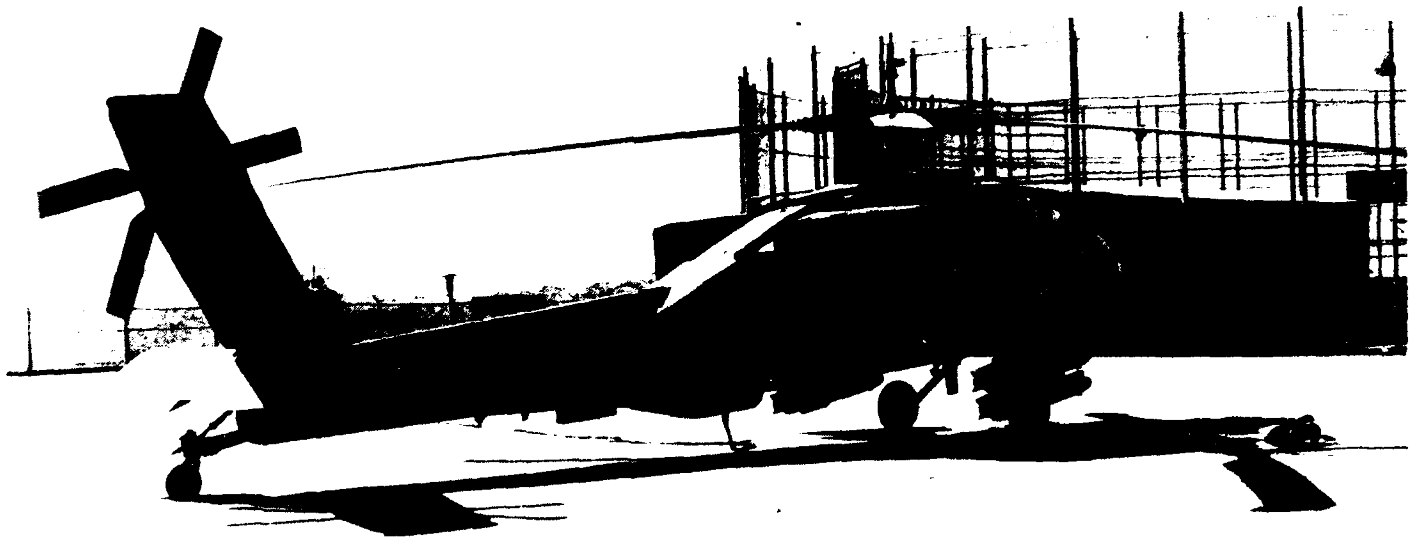


Photo 6 Right Rear Quartering View

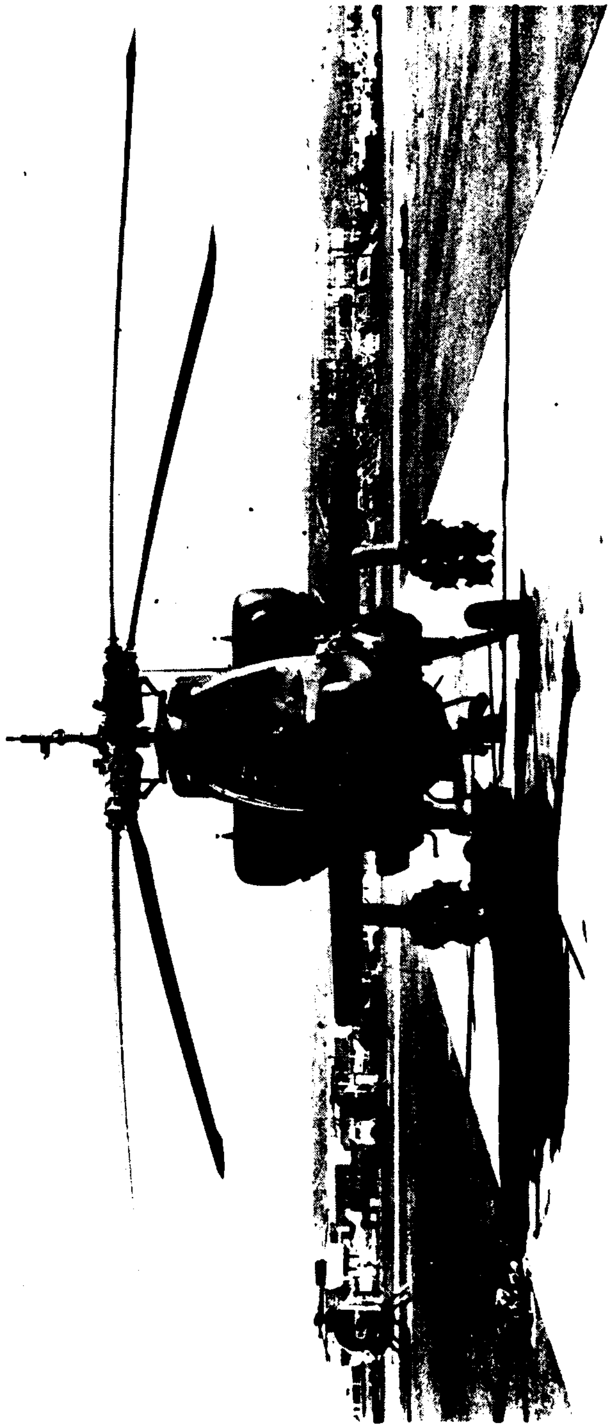


Photo 7. Front View

[illegible] ing Gear Opening Cover (Top)

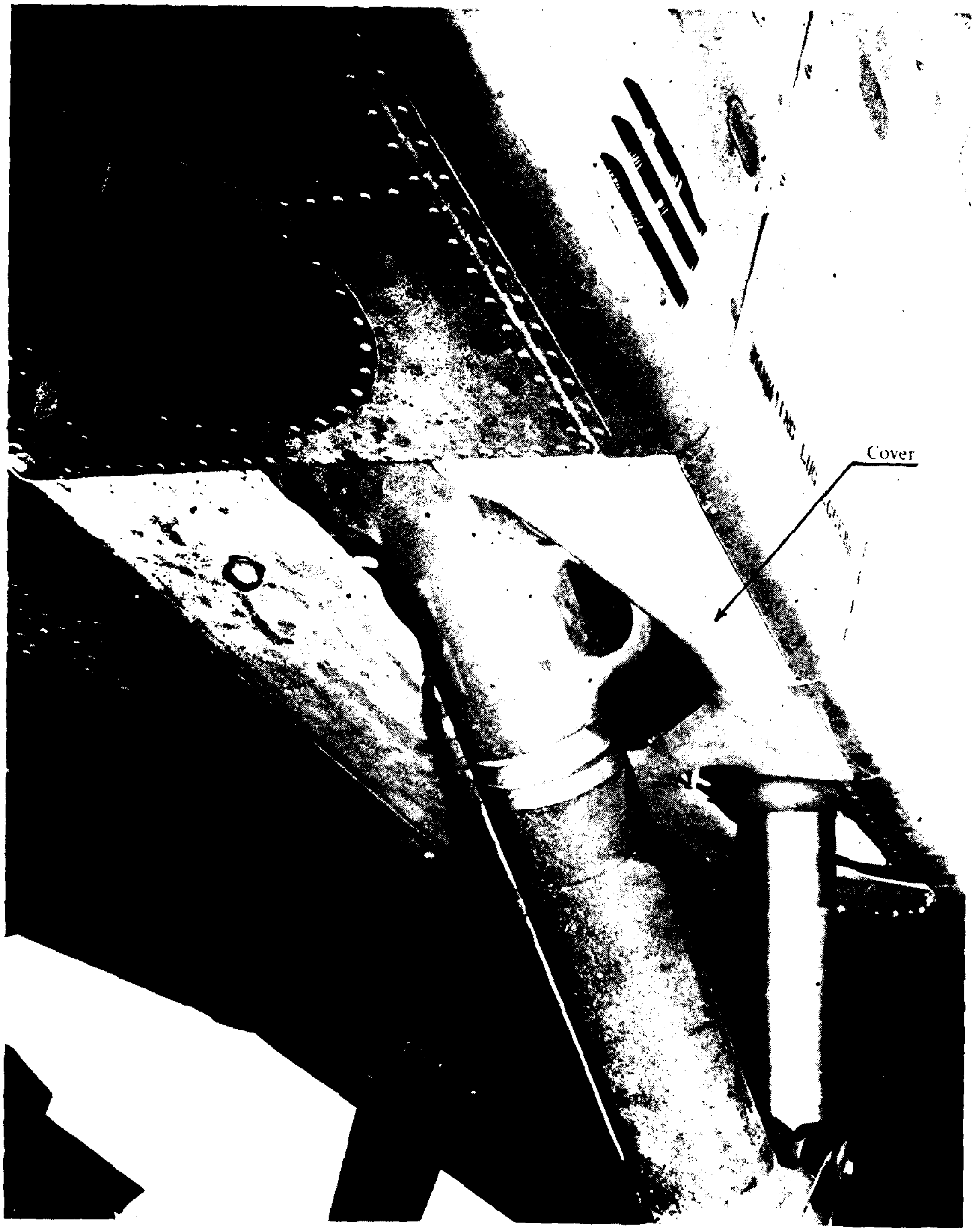


Photo 9. Main Landing Gear Opening Cover (Bottom)

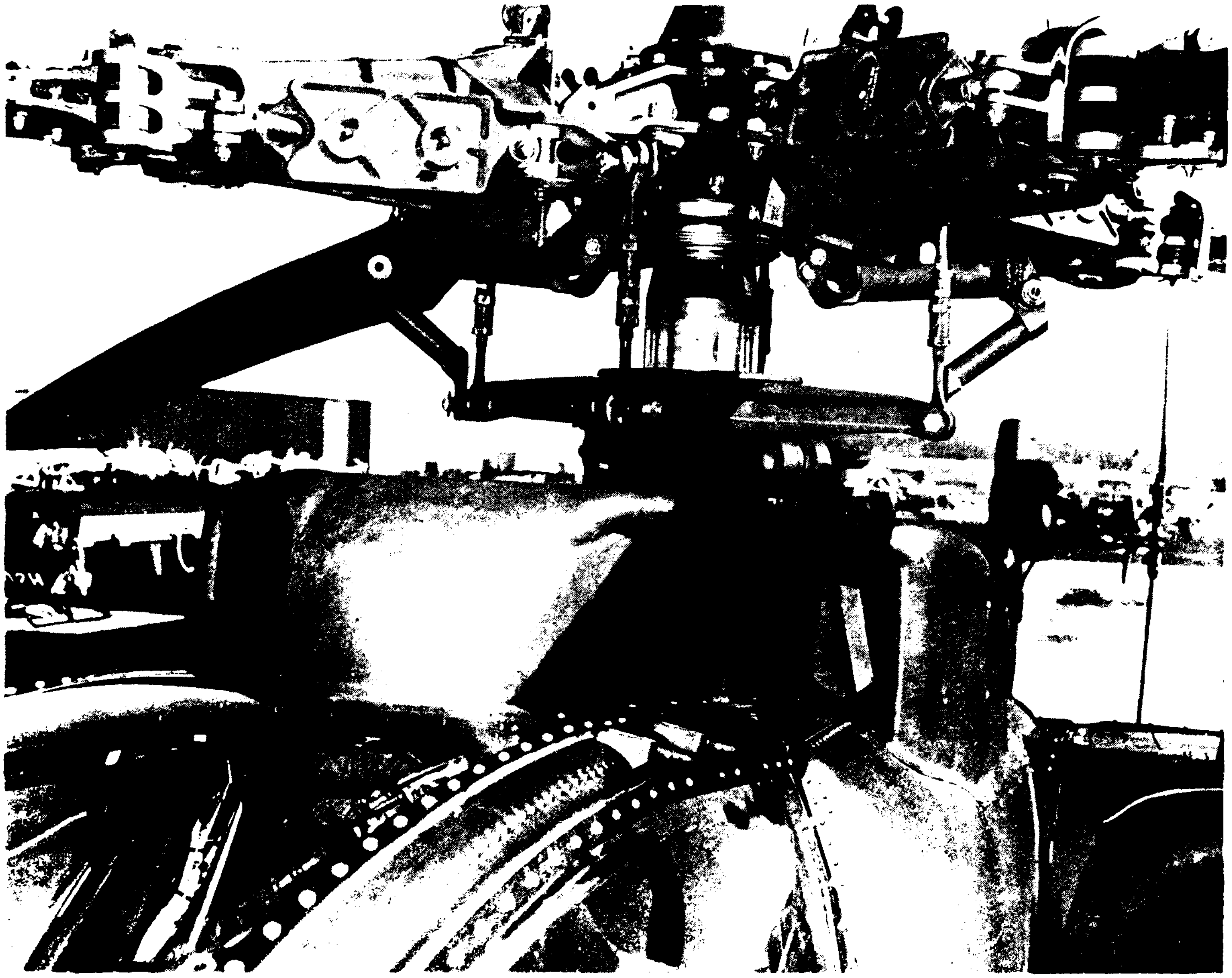


Photo 10. Main Rotor Mast Fairing

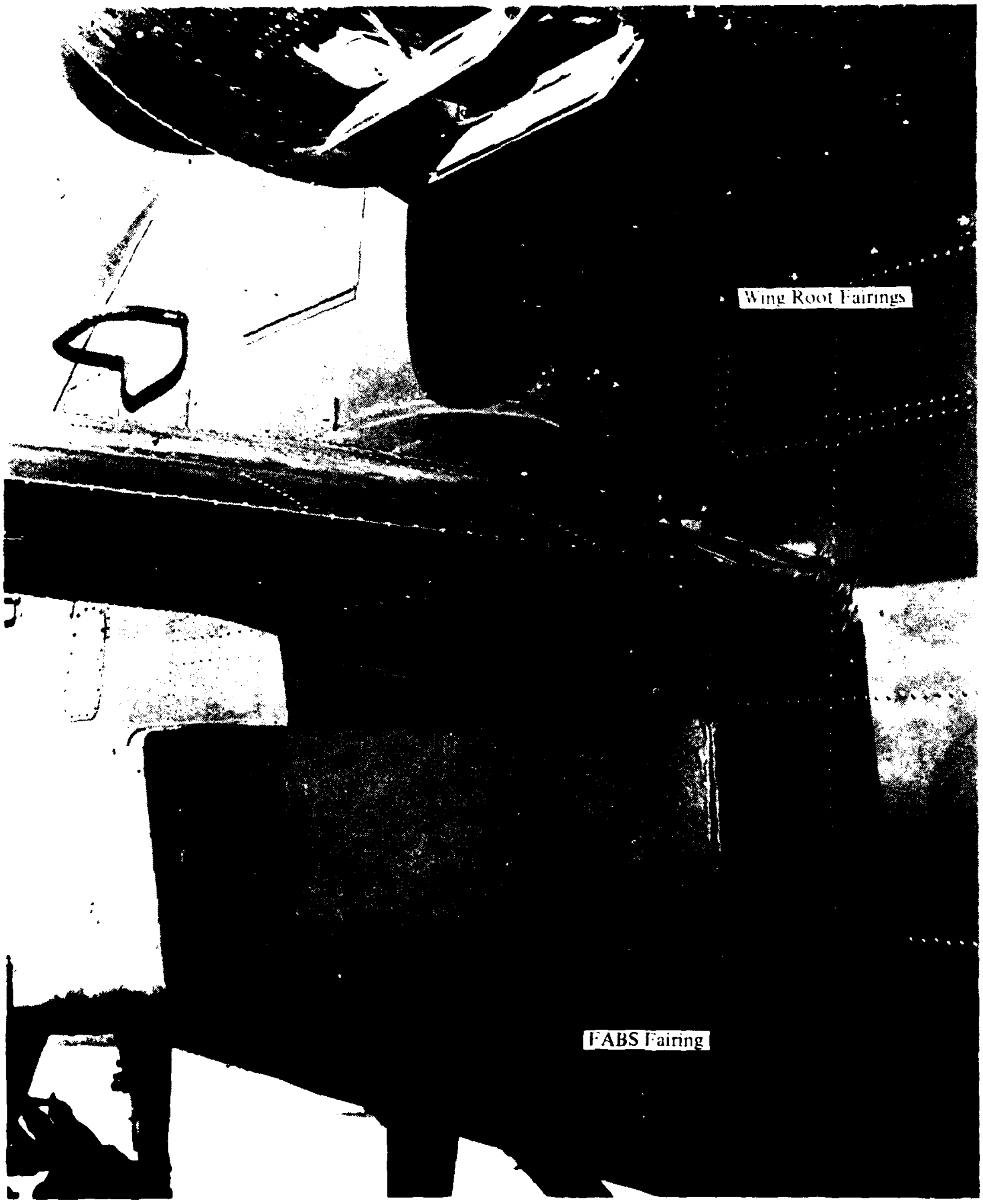


Photo 11. Wing Root and FABS Rear Extension Fairings

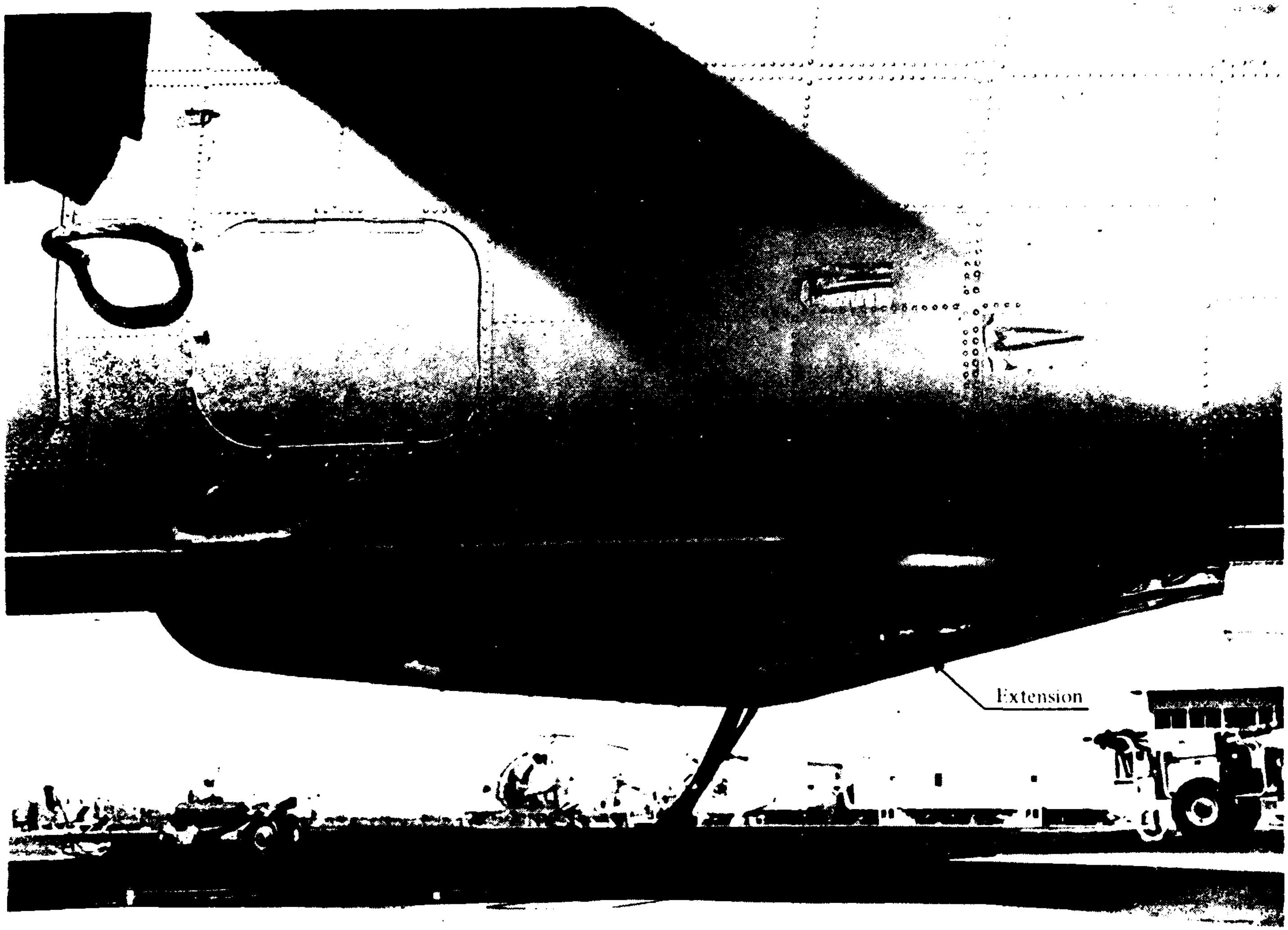


Photo 17 Doppler Radar Antenna Fairing Extension

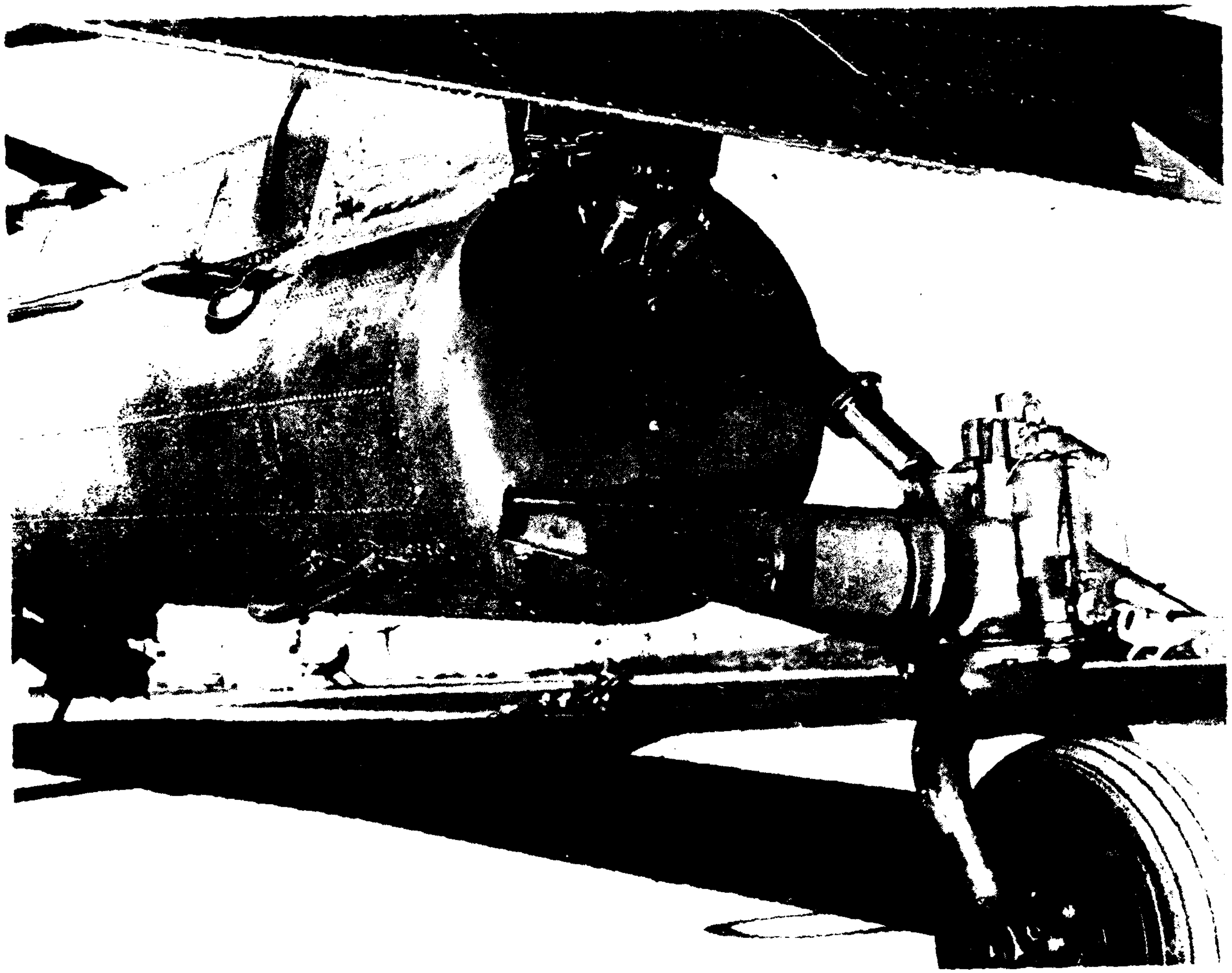


Photo 13. Closure on Rear of Tail Cone

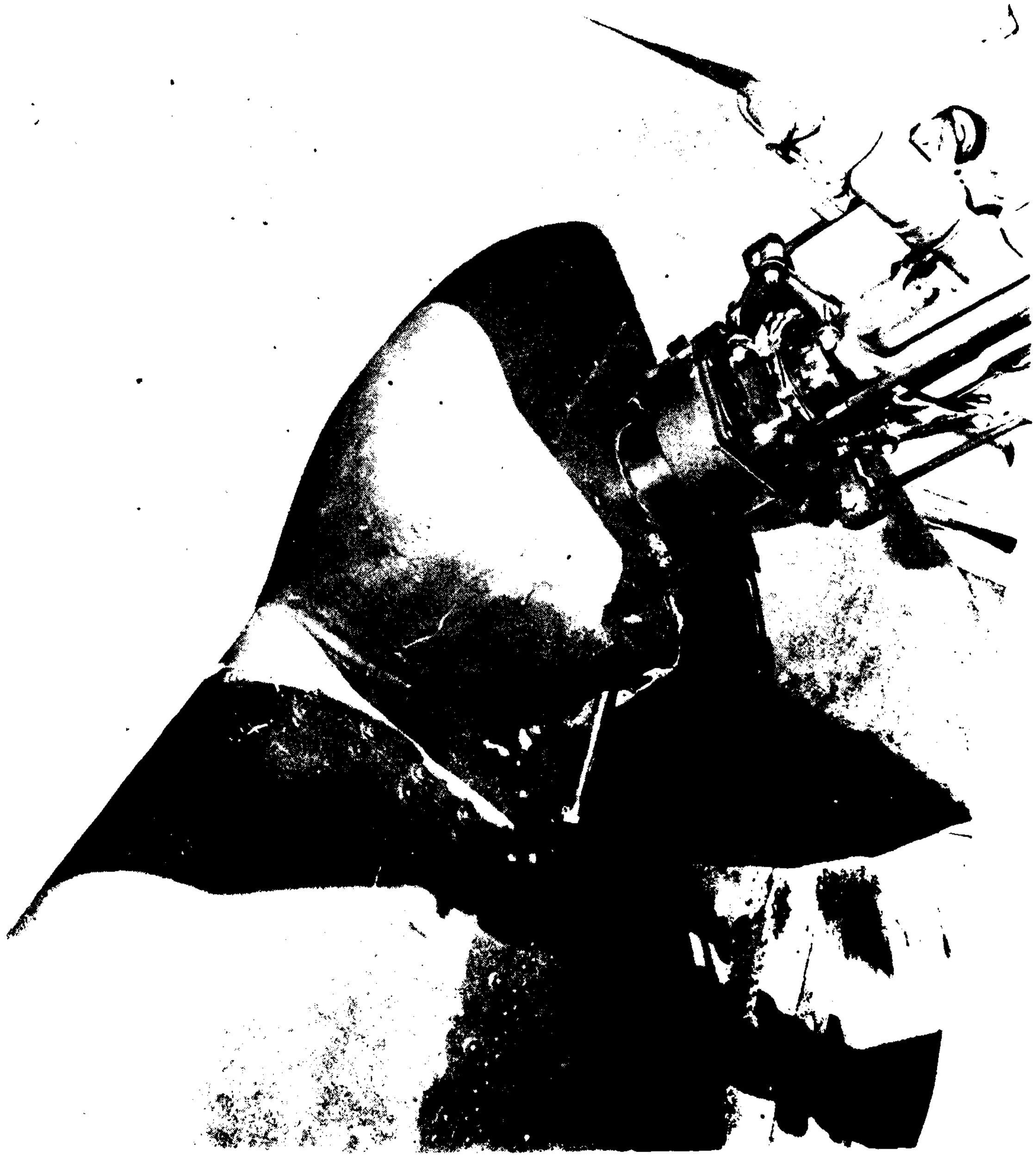
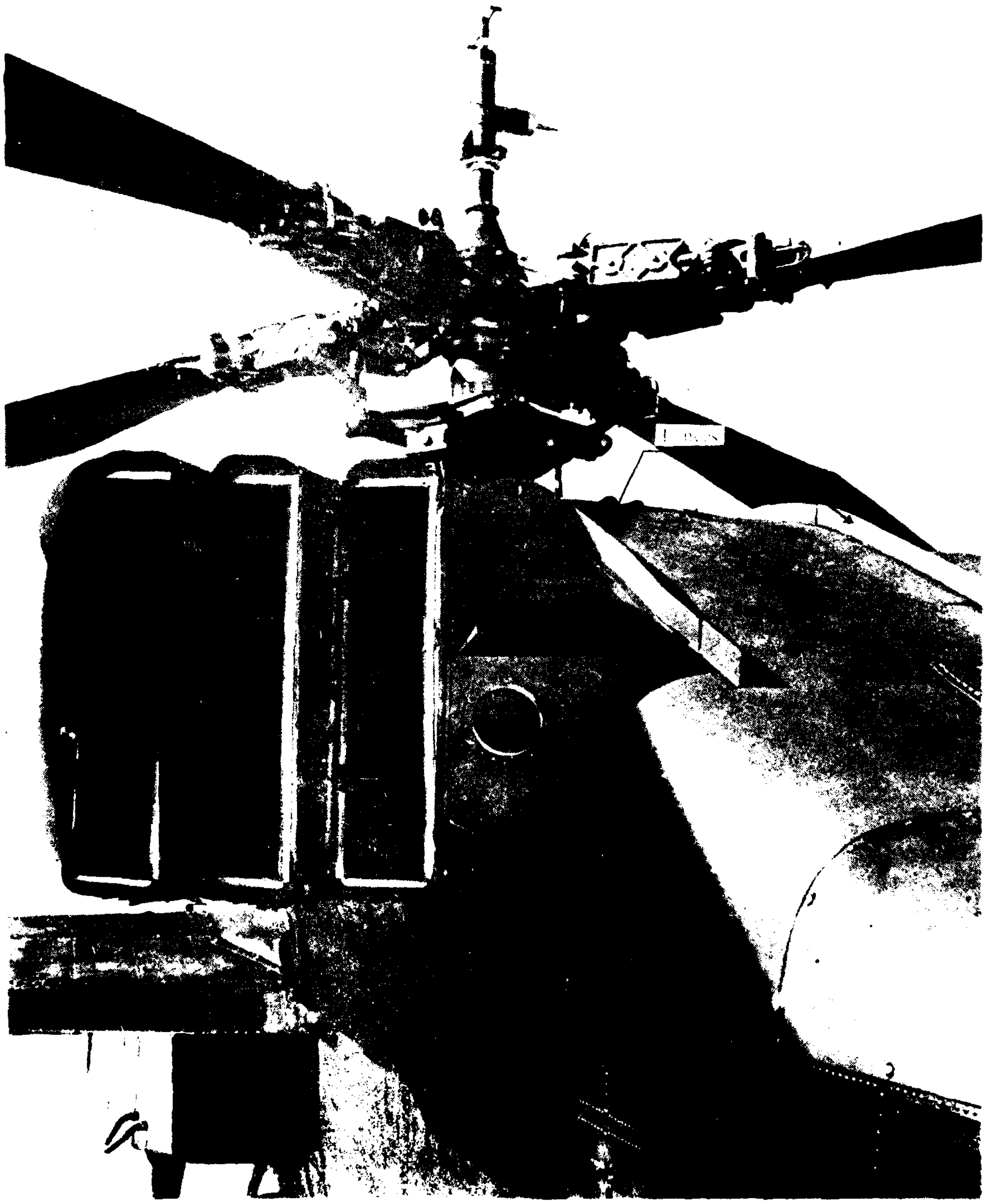


Photo 14. Tail Rotor Gear Box Failure



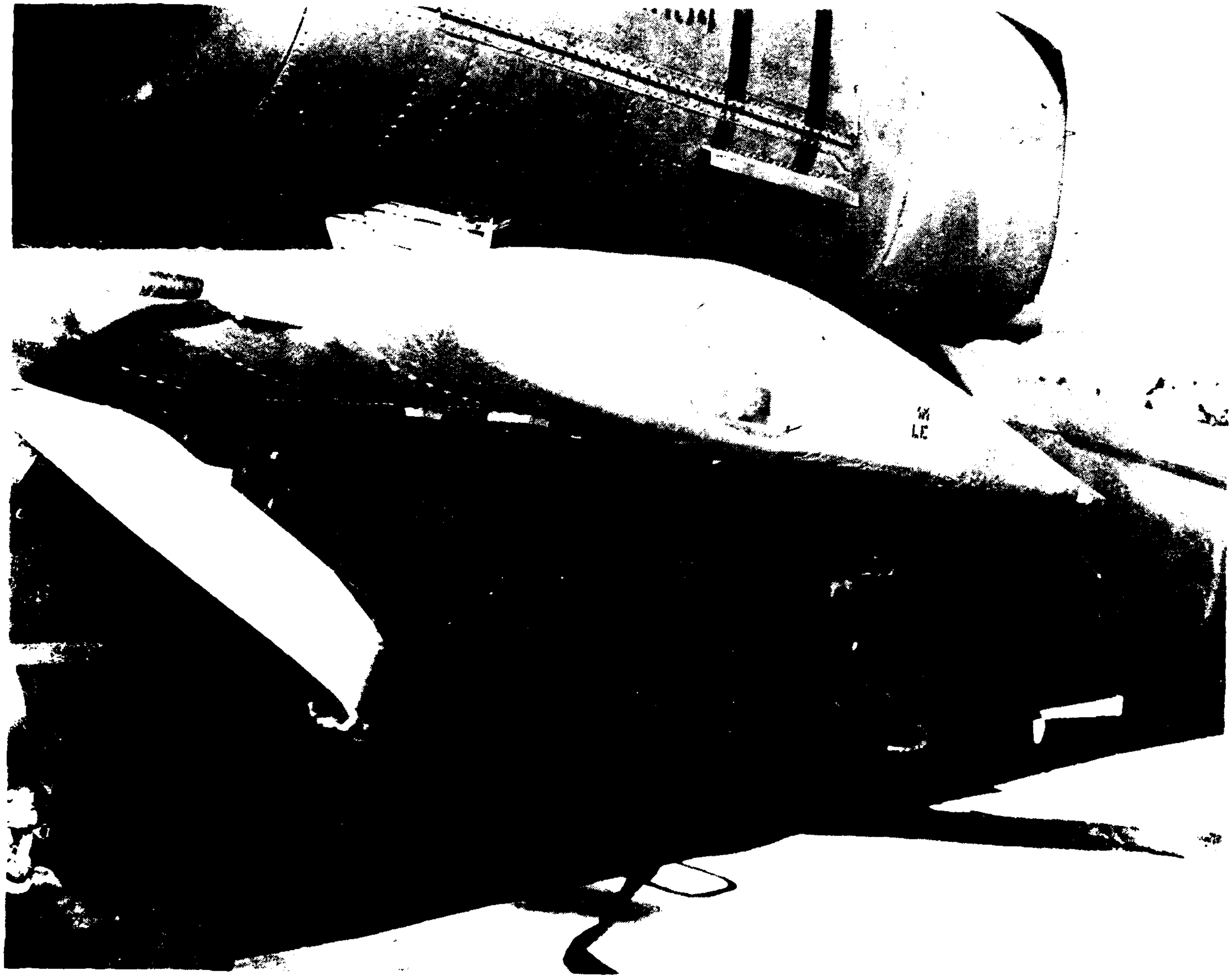


Photo 16. Wing Tip Light Fairing

GENERAL

1. The YAH-64 Advanced Attack Helicopter (fig 1) is a tandem, two-place twin turbine engine, single main rotor aircraft manufactured by Hughes Helicopters, a division of Summa Corporation. The main rotor is a four-bladed fully articulated system. It is supported by a stationary mast which transmits flight loads directly to the fuselage. The tail rotor is a four-bladed semi-rigid, delta-hinged system incorporating elastomeric teetering bearings. The rotors are driven by two General Electric YT 700-GE-700R engines through the power train shown in figure 2. An auxiliary power unit (APU) is installed primarily for starting the engines and to provide electrical and hydraulic power when the aircraft is on the ground and rotors are not turning. The aircraft is designed to carry various combinations of ordinance stores internally in the ammunition bay and externally on the four wing store positions. The YAH-64 is designed to operate during day, night and marginal weather combat conditions using the Martin Marietta Target Acquisition Designation System (TADS)/Pilot's Night Vision System (PNVS). The test aircraft, S/N 77-23258, (photos 1 through 16) was configured with an aerodynamic mockup of the TADS/PNVS, a 30mm CHAIN GUN and a HELLFIRE missile launcher loaded with four dummy missiles on each of the two inboard wing pylons. The major aircraft modifications since Engineer Design Test 2 (EDT-2) consist of an increase in tail rotor diameter, the installation of a moveable horizontal stabilator, the incorporation of Digital Automatic Stabilization Equipment (DASE), installation of a vertical vibration absorber, stiffening of the canopy window panels and fixing the position of the moveable wing flaps. Various modifications were incorporated to reduce drag and are listed in table 1. Neither the Back Up Control System (BUCS) nor the Electronic Attitude Direction Indicator (EADI) were operational during this test.

DIMENSIONS AND GENERAL DATA

	EDT-2 Mod 2B	EDT-4
Main Rotor		
Diameter (ft)	48	48
Blade chord (in.)	21.0*	21.0*
Main rotor blade area (ft^2)	166.5	166.5
Main rotor disc area (ft^2)	1809.56	1809.56
Main rotor solidity (thrust weighted, no tip loss)	0.092	0.092
Airfoil	HH-02**	HH-02**
Twist	-9 deg	-9 deg
Number of blades	4	4
Rotor speed at 100 percent N_R (rpm)	289.3	289.3
Tip speed at 100 percent N_R (ft/sec)	727.09	727.09

*Includes tips.
**Outer 20 inches swept back 20 degrees and transitioned to an NACA 006 airfoil.

	EDT-2 Mod 2	EDT-4
Tail Rotor		
Diameter (ft)	8.33	9.17
Chord constant (in.)	10	10
Tail rotor blade area (ft^2)	10	14.89
Tail rotor disc area (ft^2)	54.54	66.0
Tail rotor solidity	0.2475	0.2256
Airfoil	NACA 632-414 (modified)	NACA 632-414 (modified)
Twist (deg)	8	8.8 washout
Number of blades	4	4
Rotor speed at 100 percent N_R (rpm)	1411	1403.4
Distance from main rotor mast centerline (C_L)(ft)	28.49	29.67
Tip speed at 100 percent N_R (ft/sec)	615.44	673
Teetering angle (deg)	35	35
Horizontal Stabilizer/Stabilator		
Weight (lb)	112.8	77.3
Area (ft^2)	32.99	33.36
Span (ft)	11.46	10.67
Tip chord (ft)	1.94	2.65
Root chord (ft)	3.81***	3.60
Airfoil	NACA 0015	NACA 0018
Geometric aspect ratio	3.98	3.41
Incidence of chord line (deg)	+1	Variable
Sweepback of leading edge (deg)	0	2.89
Sweepback of trailing edge (deg)	-19.13 deg (swept forward)	-7.23 deg (swept forward)
Dihedral (deg)	0	0
Vertical Stabilizer		
Area (from boom C_L)(ft^2)	32.80	32.2
Span (from boom C_L)(in.)	113.0	113.0
Root chord (at boom C_L)(in.)	47.84	44.0
Geometric aspect ratio	2.77	2.5
Airfoil	NACA 4415 modified at root (C_L boom tapering to NACA 4416 at 66 in. from boom C_L)	NACA 4415 modified

***Reference is 3.2 inches from centerline (C_L).

	EDT-2 Mod 2	EDT-4
Leading edge sweep (deg)	0	29.4
(to 66 in. from boom C_L)	32.28	
(from 66 to 113 in. from boom C_L)	25.68	
Rudder deflection	12+10 deg tab extension (above fold joint****), Below fold joint deflection should fair from 12 deg at top to half ellipse at bottom	16 deg above W.L. 196.0 fairing below W.L. 196.0 to a half ellipse at W.L. 153.2.
Wing		
Span (ft)	16.33	16.33
Mean aerodynamic chord (in.)	45.9	45.9
Total area (ft^2)	61.59	61.59
Flap area (ft^2)	8.71	8.71 (fixed)
Airfoil at root	NACA 4418	NACA 4418

****Deleted from EDT-4

FLIGHT CONTROL DESCRIPTION

General

2. The YAH-64 helicopter employs a single hydromechanical irreversible flight control system. The hydromechanical system is mechanically activated with conventional cyclic, collective and directional pedal controls, through a series of push-pull tubes attached to four airframe-mounted hydraulic servoactuators. The four hydraulic servoactuators control longitudinal cyclic, lateral cyclic, main rotor collective and tail rotor collective pitch. Hydraulic power is supplied by two independent 3000-psi hydraulic systems which are powered by hydraulic pumps mounted on the accessory gearbox to allow full operation under a dual-engine failure condition. A Digital Automatic Stabilization Equipment (DASE) system is installed to provide rate damping. The DASE control authority is limited to 10 percent of pilot control authority in pitch, roll, and yaw. The DASE also provides attitude hold and a Hover Augmentation System (HAS). An electrically-actuated horizontal stabilator is attached to the lower aft side of the vertical stabilator. Movement of the stabilator can be controlled either manually or automatically. A Trim Feel System (TFS) is incorporated in the cyclic and pedal controls to provide a control force gradient with control displacement from a selected trim position. A trim release switch, located on the cyclic grip, provides either a momentary or continuous interruption of the TFS in all axes simultaneously to allow the cyclic or pedal controls to be placed in a new trim position. Full control travel is 10.2 inches in the longitudinal control, 8.9 inches in the lateral control,

Table 1. Drag Reduction Modifications

ECRR* Number	Date	Description
39319	7 Aug 80	Addition of main landing gear opening covers on top and bottom of forward avionics bays (FABS)
39320	7 Aug 80	Trailing edge of vertical stabilizer adjusted to 16 degrees
40986	7 Aug 80	Addition of upper main rotor mast fairing
40990	7 Aug 80	Addition of FABS rear extension fairings
40991	7 Aug 80	Doppler radar antenna fairing extension
40992	7 Aug 80	Addition of wing root fairings
40988	7 Aug 80	Closure on rear end of tail cone
53296	19 Sep 80	Improved tail rotor gear box fairing
53927	19 Sep 80	Improved fairing between engine nacelles
53501	7 Aug 80	Addition of fences on fairing between engine nacelles
53298	3 Nov 80	Wing store flight stow position adjusted up 3 degrees
39322	7 Aug 80	Addition of wing tip light fairing

*Engineering change request and record

12.0 inches in the collective control and 5.4 inches in the directional pedals (during the test program, the left pedal stop was moved 1.2 inches, restricting total travel to 4.2 inches).

Cyclic Control System

3. The cyclic control system (fig 3) consists of dual-tandem cyclic controls attached to individual support assemblies in each cockpit. The support assembly houses the primary longitudinal and lateral control stops, and two linear variable displacement transducers (LVDT) designed to measure electrically the longitudinal and lateral motions of the cyclic for DASE computer inputs. A series of push-pull tubes and bellcranks transmits the motion of the cyclic control to the servoactuators and the mixer assembly. Motion of the mixer assembly positions the nonrotating swashplate, which transmits the control inputs to the rotating swashplate to control the main rotor blades in cyclic pitch (fig 4). The cyclic stick grips are shown in figure 5. A stick fold linkage is provided to allow the copilot/gunner (CPG) to lower the cyclic stick to prevent interference when operating the weapon systems.

Collective Control System

4. The collective pitch control system (fig 6) consists of dual-tandem controls which transmit collective control inputs to the main rotor through a series of push-pull tubes and bellcranks attached to the collective servoactuator. Motion of the servoactuator is transmitted through the mixer assembly to the swashplate to control the main rotor blades in collective pitch. Collective inputs are also transmitted to the load demand spindle of each engine hydromechanical unit (HMU). The HMU meters the fuel as appropriate to provide collective pitch compensation. Located at each collective control base assembly are the primary control stop, an LVDT, and a 1g balance spring. The LVDT supplies electrical inputs to the stabilator control units.

5. The collective control stick (fig 7) incorporates a switch box assembly, an engine chop collar, a stabilator control panel and an adjustable friction control. The engine chop collar allows rapid deceleration of the engine to flight idle, primarily to allow immediate action in the event of a tail rotor failure.

Directional Control System

6. The directional control system (fig 8) consists of a series of push-pull tubes and bellcranks which transmits directional pedal inputs to the tail rotor hydraulic servoactuator located in the vertical stabilizer. Attached to each directional pedal assembly are the primary tail rotor control stops and one LVDT. Two sets of wheel brake cylinders are attached to the directional pedals and a 360 degrees swiveling tail wheel is incorporated. The tail wheel may be locked in the trailing position by means of a switch located on the pilot's instrument panel.

Trim Feel System

7. A TFS is incorporated in the longitudinal, lateral, and directional control systems. The TFS uses individual magnetic brake clutch assemblies in each of the control linkages. Trim feel springs are incorporated to provide a control force gradient and positive control centering. The electromagnetic brake clutch is powered by 28 VDC and is protected by the trim circuit breaker. A complete DC electrical

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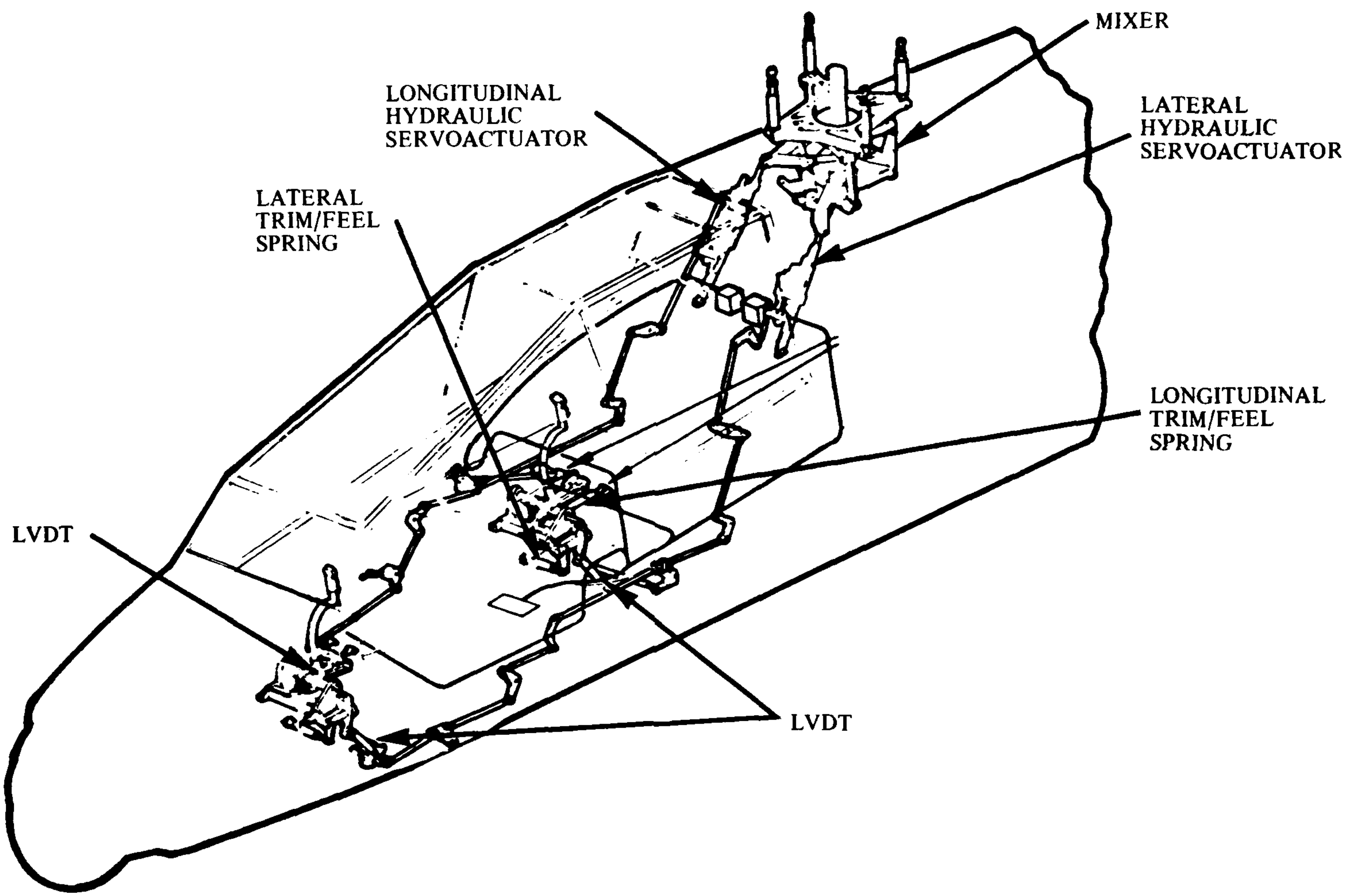


Figure 3. Cyclic Control System

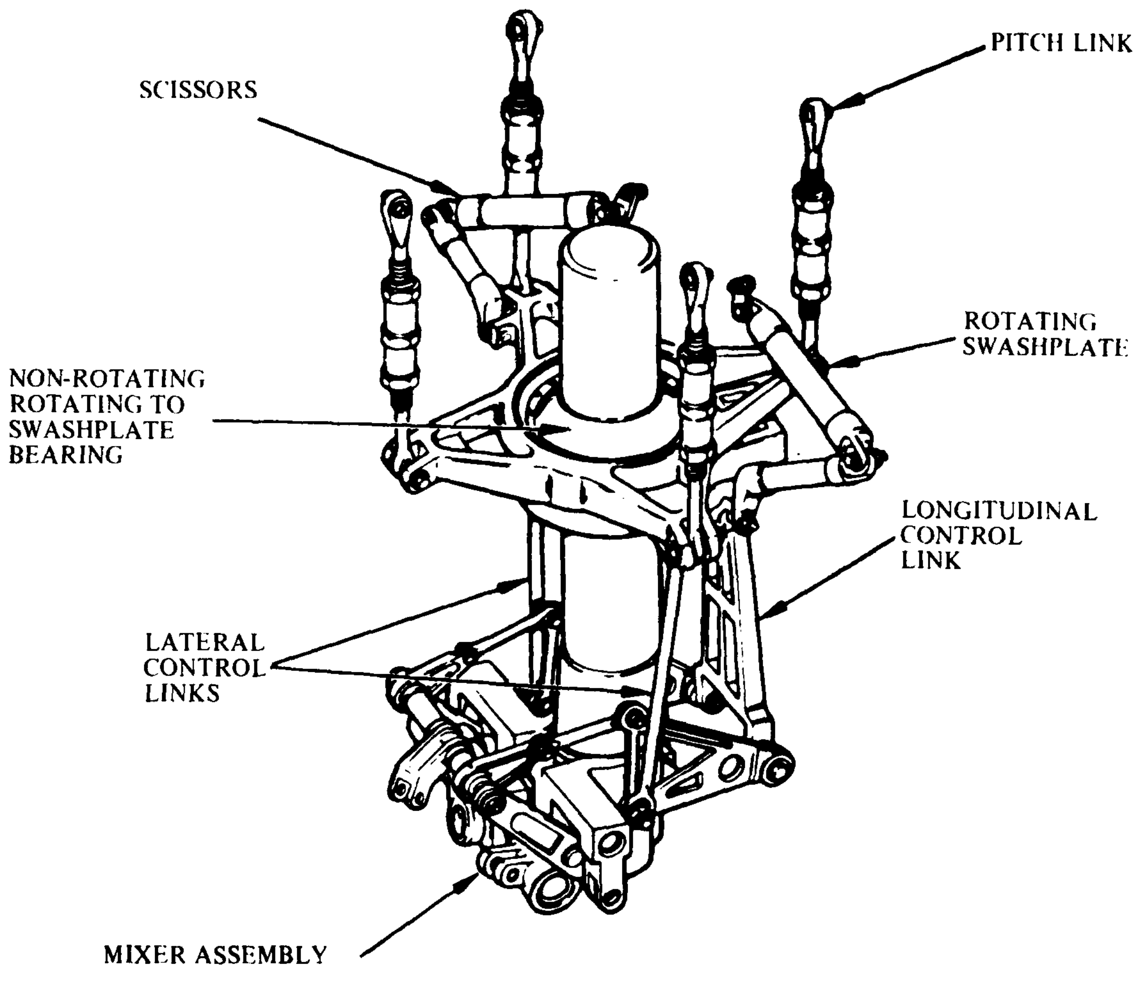
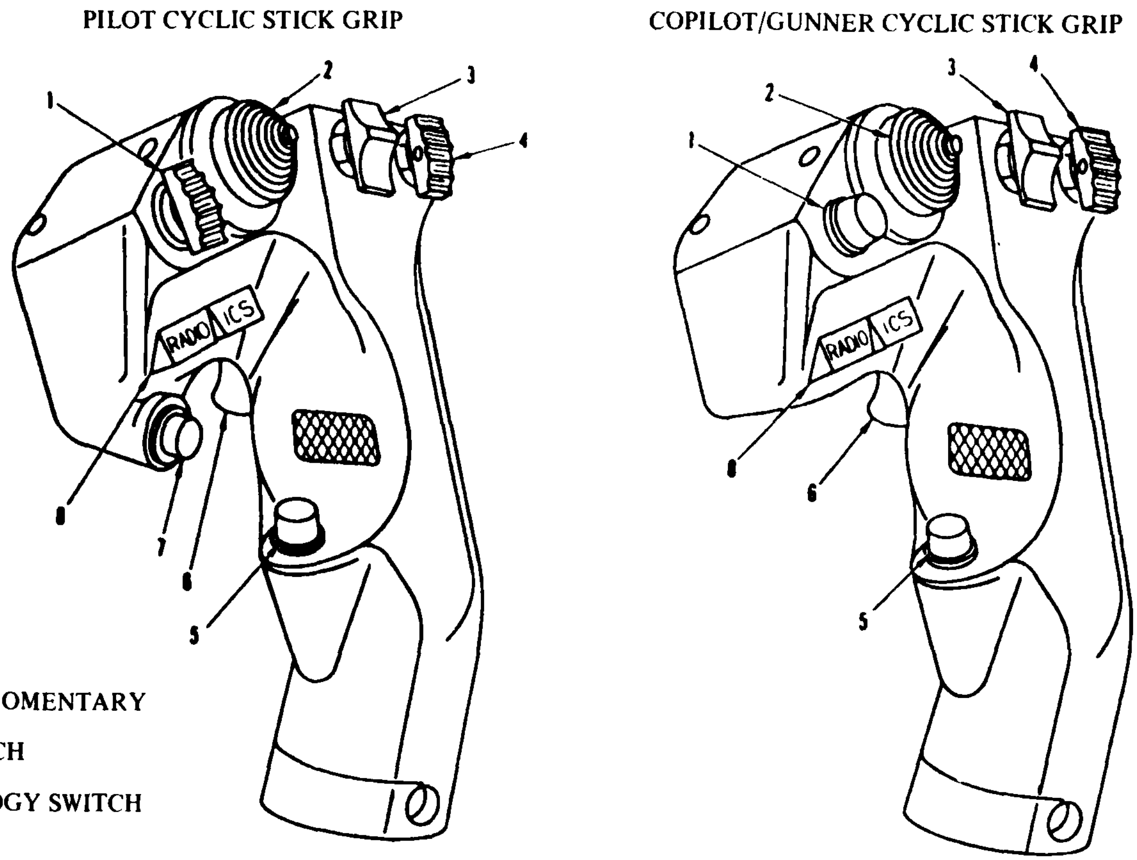


Figure 4. Main Rotor Swashplate Assembly



1. TRIM RELEASE SWITCH
   CPG GRIP-MOMENTARY
   PILOT GRIP-DISCRETE/MOMENTARY
2. WEAPONS ACTION SWITCH
3. FLIGHT MODE SYMBOLOGY SWITCH
4. TO BE DETERMINED
5. DASE RELEASE SWITCH
6. GUARDED TRIGGER SWITCH
7. REMOTE TRANSMITTER SELECTOR SWITCH
   (PILOT GRIP ONLY)
8. RADIO, ICS ROCKER SWITCH

Figure 5. Cyclic Stick Grips

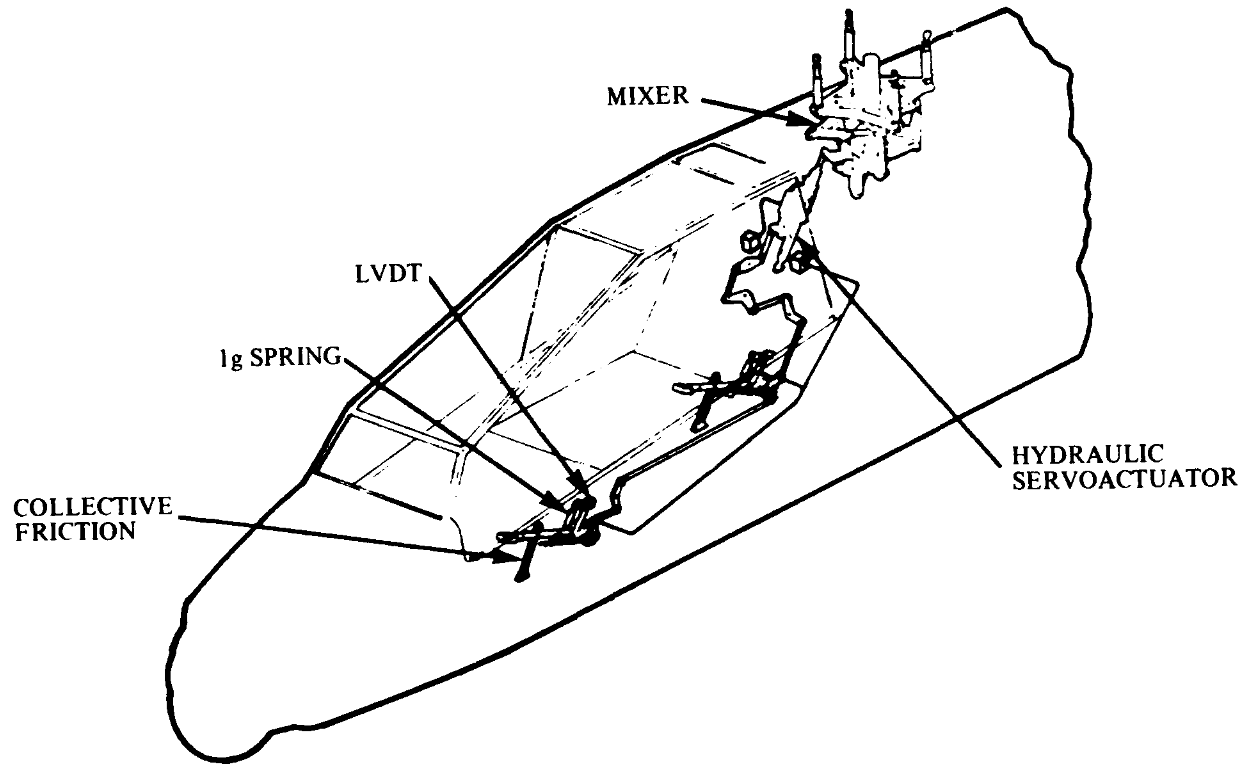


Figure 6. Collective Control Subsystem

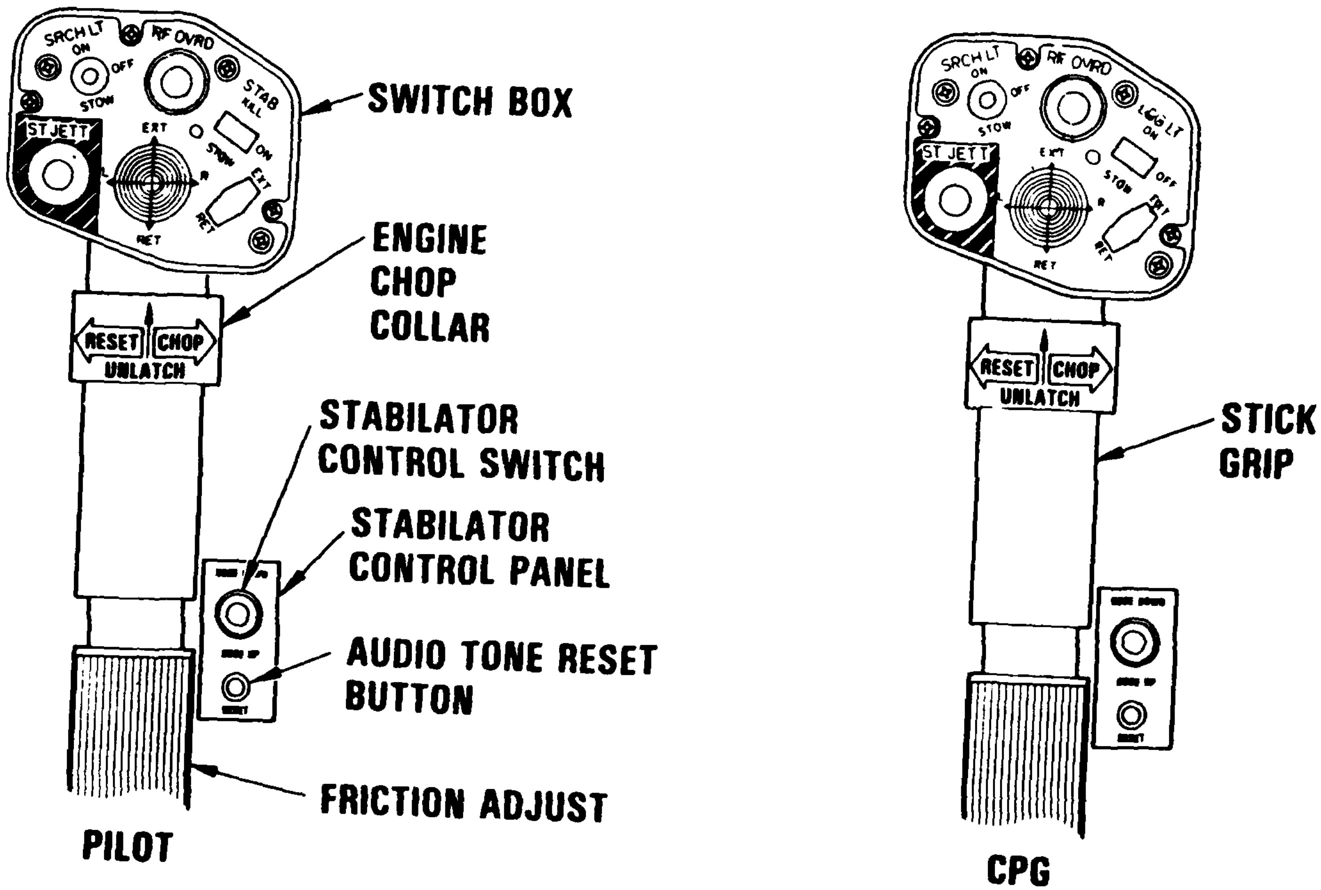


Figure 7. Collective Stick and Switch Box

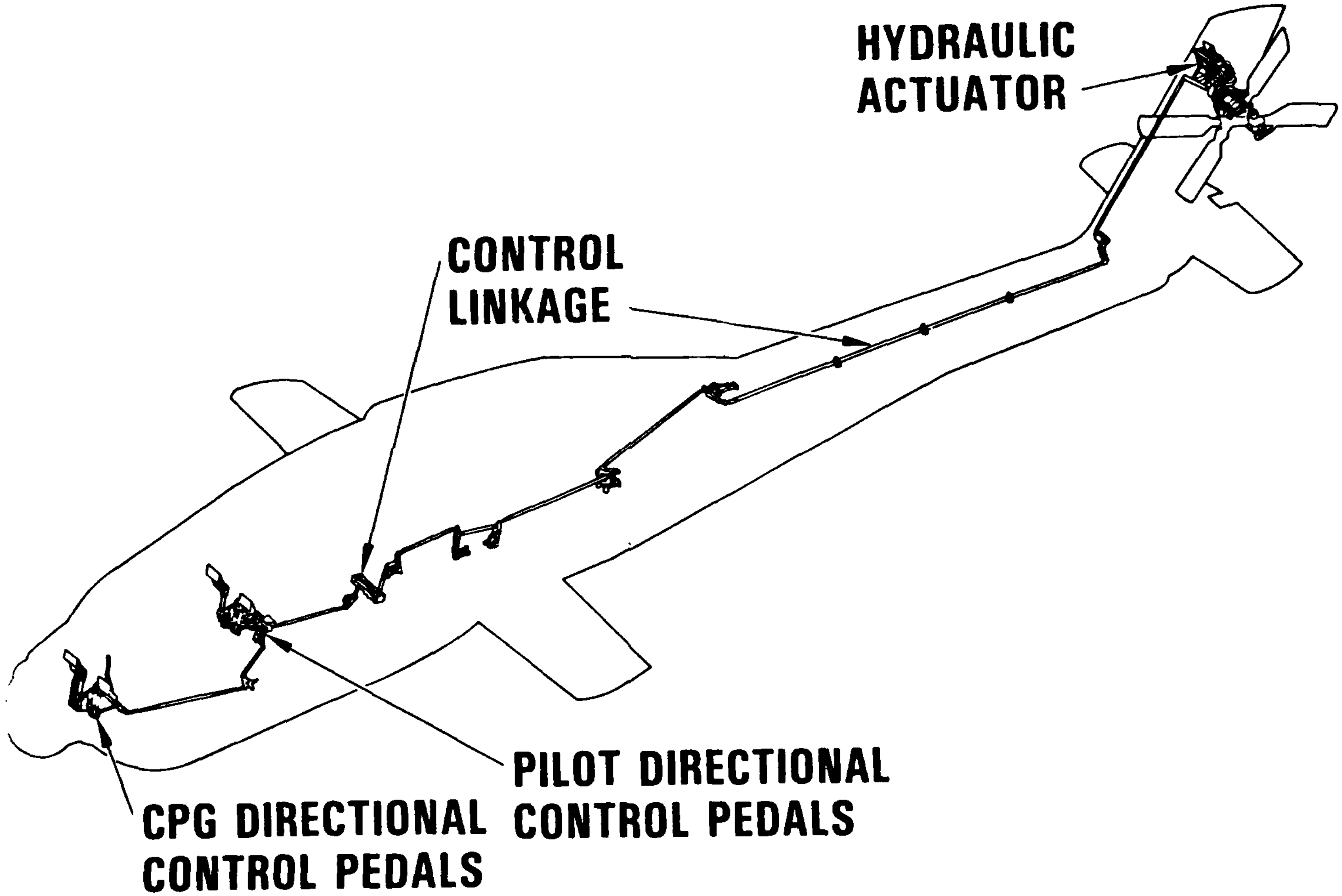


Figure 8. Directional Controls

failure will disable the TFS and allow the cyclic and directional pedals to move freely without resistance from the trim feel springs. The trim release switch on the pilot's cyclic grip allows either momentary or discreet release of the TFS. The CPG has a momentary release capability only.

Horizontal Stabilator

8. The horizontal stabilator is attached to the aft lower portion of the vertical stabilizer. The stabilator airfoil has a span of 10.7 feet, the tip cord is 2.7 feet and the root cord is 3.6 feet (total 33.4 square feet). A dual, series 28 VDC electromechanical actuator allows incidence changes of +45 degrees (trailing edge down) to -10 degrees of travel. Software limits in the stabilator control unit (SCU) limits the incidence change to +25 degrees and -5 degrees in the automatic mode of operation. Automatic positioning of the stabilator during flight is primarily a function of airspeed and collective position as shown in figure 9. The stabilator also responds with low gain (0.25 deg/deg/sec) and limited authority (±2.5 deg) to pitch rate inputs to the SCU. Safety features include an automatic shutdown capability which allows operation in the manual mode by means of a stabilator control panel located on each collective stick. An audio step tone is associated with the failure of the automatic mode of operation. A stabilator kill switch, located on the pilot's collective stick, disables both the automatic and manual operation to protect against a hardover failure.

9. The stabilator is controlled in the automatic mode by two SCUs. Each SCU controls one side of the dual actuator. Both SCUs receive collective control position information from an LVDT. Two independent pitch rate gyros provide pitch rate information to the SCUs (one gyro for each SCU). The Air Data System (ADS) provides airspeed to both SCUs. The left-hand pitot-static system supplies airspeed to one SCU and the right-hand system provides airspeed to the other SCU. Both SCUs receive position information from both sides of the dual actuator. The maximum rate of stabilator travel is 8 degrees per second.

10. The SCUs have a fault detection feature which will switch the stabilator mode of operation from automatic to manual if any of the following conditions are sensed:

a. A mismatch between the positions of the two halves of the actuator equivalent to 10 degrees of stabilator travel. If there is a runaway failure of one side of the actuator, this feature will disable the automatic mode after 10 degrees of stabilator travel.

b. A mismatch in the rates of actuator travel of more than 10 degrees per second. This could only occur if one side of the actuator were extending while the other was retracting.

c. The stabilator at a position of 20 degrees or greater with an airspeed greater than 110 knots.

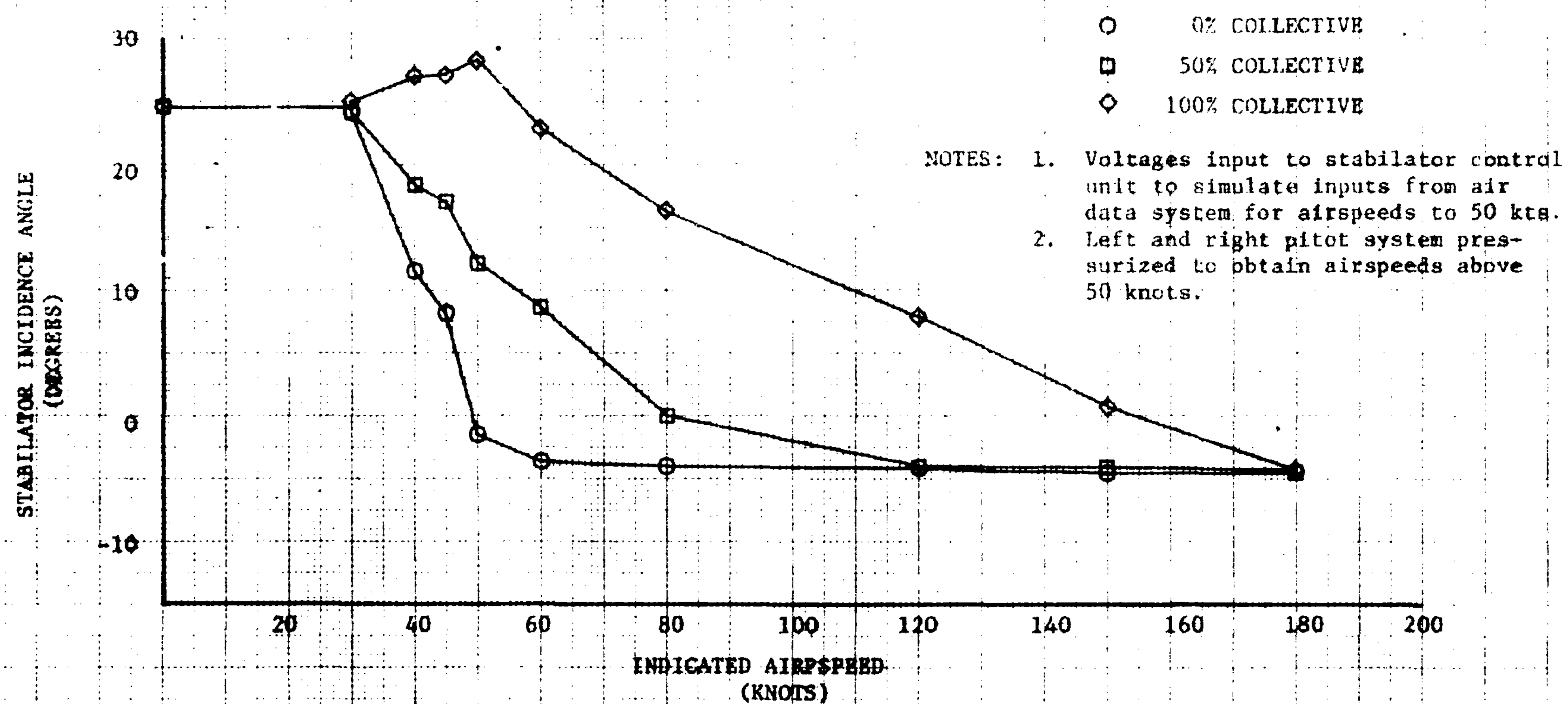
d. A stabilator angle of 30 degrees or greater with airspeed less than 60 knots.

e. Improper AC voltage or an actuator short circuit.

FIGURE 9

STABILATOR SCHEDULE

YAH-64 USA S/N 77-23258



Flight Control Rigging

11. A flight controls rigging check was performed in accordance with procedures outlined in HH Experimental Test Procedure (ETP) 7-211500000, dated 1 August 1979, (main and tail rotor controls) and ETP 7-211123600, dated 21 April 1980 (horizontal stabilator). Prior to the 18th flight of EDT-4, the tail rotor rigging was revised to conform with a HH Internal Communication document (File No. 80-1331-087), dated 20 November 1980. Horizontal stabilator rigging is shown in figure 9. Tables 2 and 3 present the collective and cyclic rigging.

12. Tail rotor rigging is shown below:

Full right pedal = 13.1 degrees (thrust to left)
Full left pedal (initial) = 33.1 degrees (thrust to right)
Full left pedal (revised) = 27.3 degrees (thrust to right)

Digital Automatic Stabilization Equipment

13. The DASE provides rate damping in three axes, control augmentation for cyclic and directional controls, pitch and roll attitude-hold in forward flight, and hover augmentation. Additionally, in forward flight the DASE attempts to maintain zero sideslip. A block diagram of the DASE is shown in figure 10. The DASE is controlled by the digital automation stabilization equipment computer (DASEC). The DASEC receives information which describes the aircraft state from the heading and attitude reference system (HARS). This information includes aircraft angular velocities (3-axes), aircraft attitudes (pitch and roll), and inertial horizontal velocity (measured by the Doppler radar). The LVDT provides longitudinal, lateral, and directional control position information to the DASEC. The DASE design calls for sideslip and airspeed information to be supplied to the DASEC by the ADS mounted on top of the main rotor mast. On the test aircraft, however, sideslip information was supplied by a transducer mounted on the test airspeed boom. The DASEC processes the information and made control inputs through the stability and control augmentation system (SCAS) solenoid valves on the longitudinal, lateral and directional servoactuators. The DASE authority is limited to ±10 percent of the pilot's control authority in each axis. Transducers on the servoactuators provide position feedback information to the DASEC.

14. The directional, lateral, and longitudinal rate damping and control augmentation functions of the DASE are activated using the three switches on the DASE control panel (fig 11) labeled YAW, ROLL, and PITCH respectively. Schematic diagrams showing gains and transfer functions are shown in figures 12 through 14. Additionally, when the YAW switch is engaged and aircraft is at an airspeed greater than 50 knots, the DASE will attempt to maintain zero sideslip.

15. Attitude-hold mode or hover augmentation system (HAS) mode may be engaged using the DASE panel switch labeled ATTD/HOVER HOLD (fig 11). At airspeeds above 50 knots, activation of this switch will engage pitch and roll attitude hold. Figures 15 and 16 are schematic diagrams describing this mode. At airspeeds below 50 knots, the HAS mode is engaged. In this mode, the DASE will attempt to maintain a position over the ground as well as maintaining pitch and roll attitudes.

Table 2. Angle Measurements
Pilots Collective and Cyclic Controls

Blade Azimuth Position (deg)	Item	Rig Pins			Stick Position			Measured Clinometer Angle (deg - min)	Leading Edge Up or Down
		Collective	Longitudinal Cyclic	Lateral Cyclic	Collective	Longitudinal	Lateral		
$\psi = 90$	1	In	In	In	Rig	Rig	Rig	0° - 15'	Down
	2	In	Out	In	Rig	Fwd	Rig	21° - 5'	Down
	3	In	Out	In	Rig	Aft	Rig	11° - 0'	Up
	4	In	In	In	Rig	Rig	Rig	0° - 20'	Down
	5	Out	In	In	Up	Rig	Rig	10° - 10'	Up
	6	Out	In	In	Down	Rig	Rig	8° - 55'	Down
	7	In	In	In	Rig	Rig	Rig	0° - 0'	Down
$\psi = 270$	8	In	In	In	Rig	Rig	Rig	1° - 0'	Down
	9	In	Out	In	Rig	Fwd	Rig	20° - 25'	Up
	10	In	Out	In	Rig	Aft	Rig	11° - 40'	Down
	11	In	In	In	Rig	Rig	Rig	1° - 0'	Down
$\psi = 0$	12	In	In	In	Rig	Rig	Rig	1° - 15'	Down
	13	In	In	Out	Rig	Rig	Left	10° - 30'	Up
	14	In	In	Out	Rig	Rig	Right	8° - 15'	Down
	15	In	In	In	Rig	Rig	Rig	1° - 10'	Down
$\psi = 180$	16	In	In	In	Rig	Rig	Rig	0° - 10'	In
	17	In	In	Out	Rig	Rig	Left	11° - 50'	Down
	18	In	In	Out	Rig	Rig	Right	7° - 50'	Up
	19	In	In	In	Rig	Rig	Rig	0° - 10'	Up

Table 3. Computation of Blade Angle Travel
Pilots Collective and Cyclic Controls

Computation	Travel (deg - min)	Tolerance (deg)
LONGITUDINAL CYCLIC		
1. Forward 1/2 (Item 9 - 2) = (If Item 2 is leading edge down, add Item 2)	20° - 45'	20° (minimum)
2. Aft 1/2 (Item 3 - Item 10) = (If Item 10 is leading edge down, add Item 10)	11° - 20'	10° (minimum)
LATERAL CYCLIC		
3. Left 1/2 (Item 13 - Item 17) = (If Item 17 is leading edge down, add Item 17)	11° - 10'	10.5° (minimum)
4. Right 1/2 (Item 18 - Item 14) = (If Item 14 is leading edge down, add Item 14)	8° - 2'	7.0° (minimum)
COLLECTIVE		
5. Full pitch travel (Item 5 - Item 6) = (If Item 6 is leading edge down, add Item 6)	19° - 5'	18.0° (minimum)
6. Collective pitch full down Measured @ 3/4 radius (Theo. chord line)		-1° to +2°
Measured @ pitch housing (Bolt pad machined surface 2.4 inches inboard of lead-lag hinge)	8° - 55'	-10° to -7°

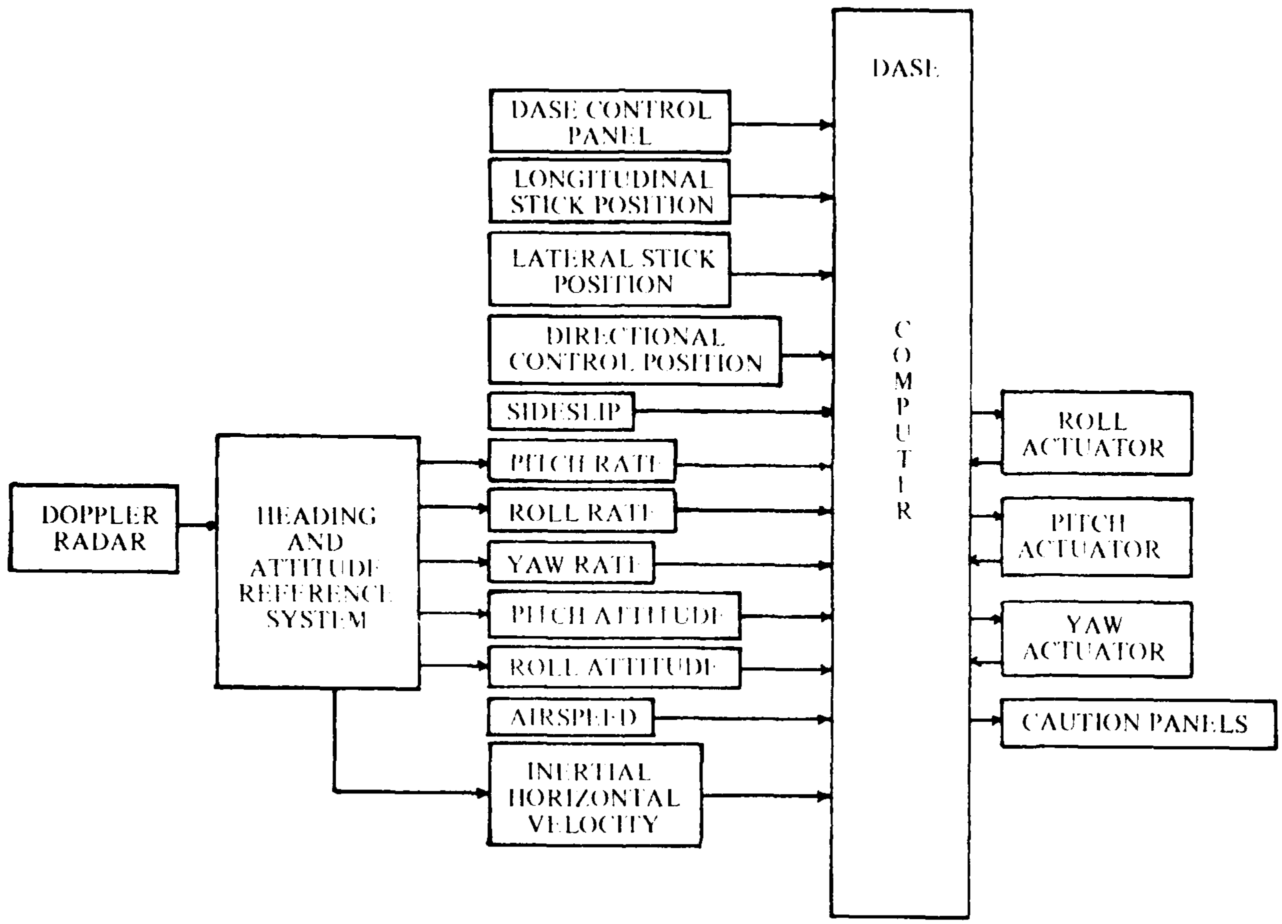


Figure 10. Digital Automatic Stabilization Equipment Block Diagram

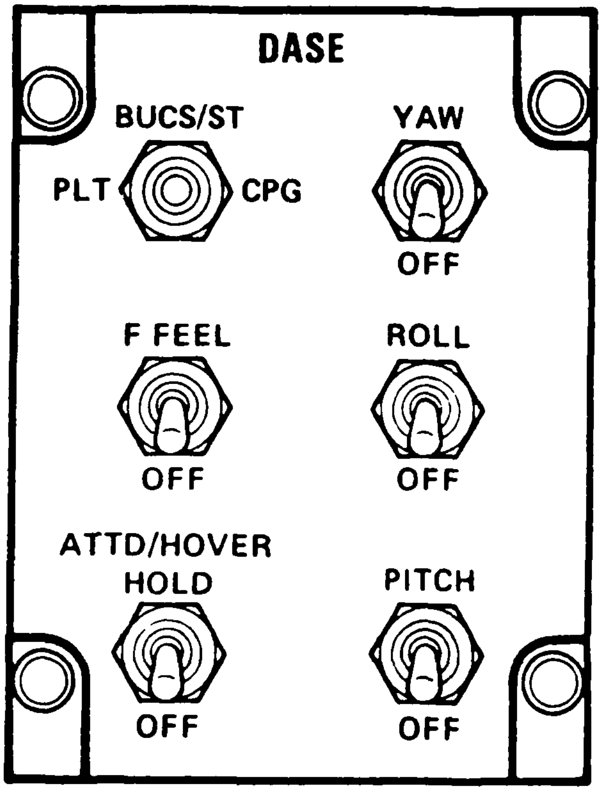


Figure 11. DASE Control Panel

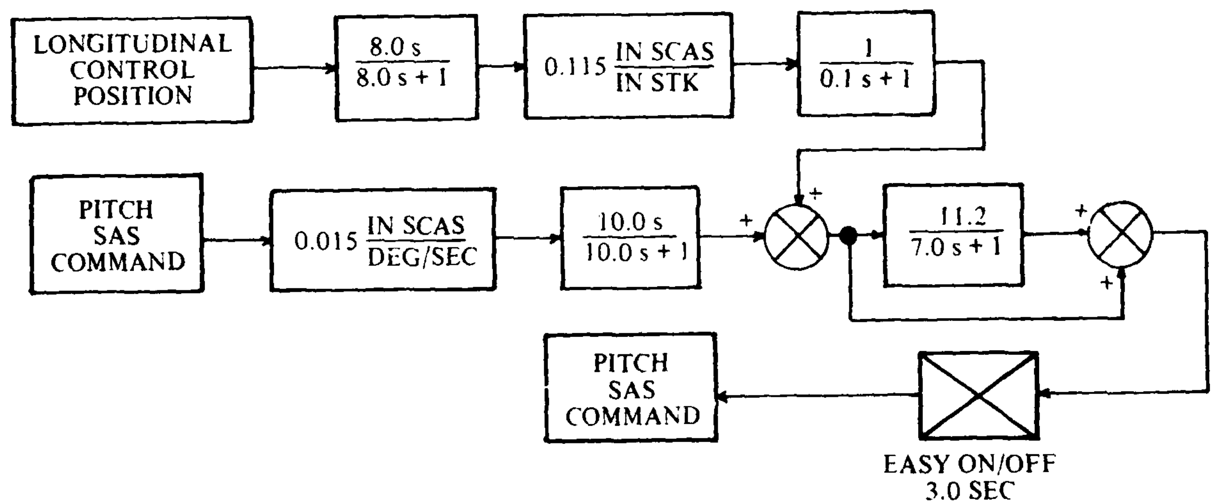


Figure 12. Pitch Rate Damping and Control Augmentation Functions

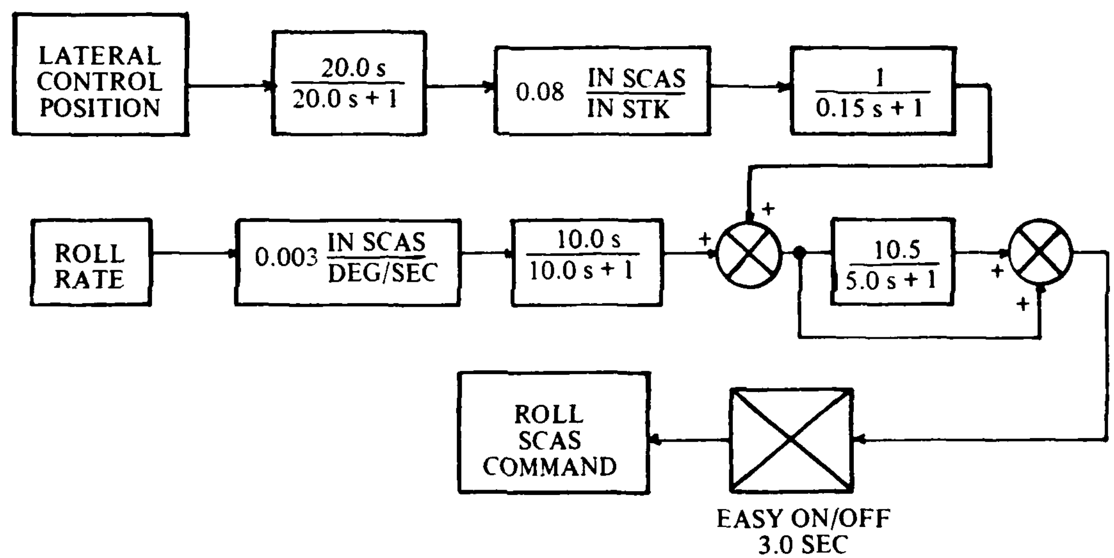


Figure 13. Roll Rate Damping and Control Augmentation Functions

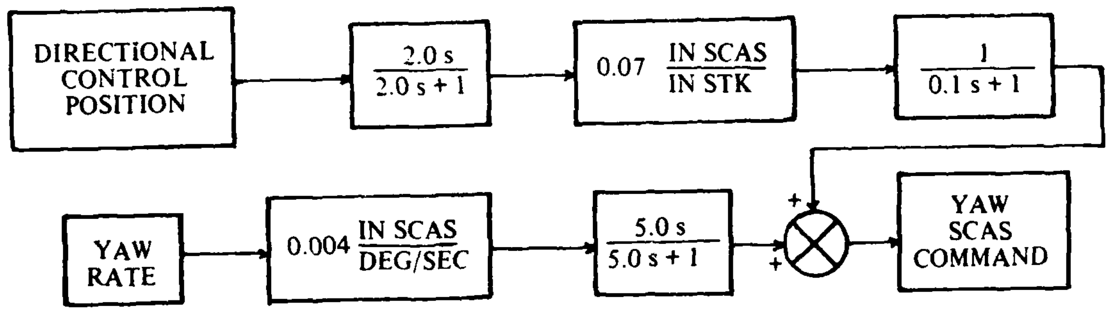


Figure 14. Yaw Rate Damping and Control Augmentation Functions

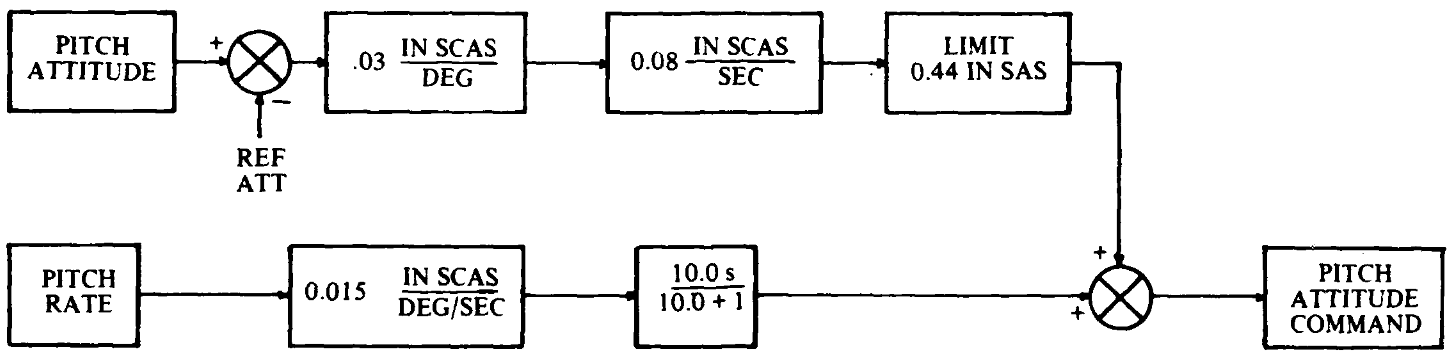


Figure 15. Pitch Attitude Hold

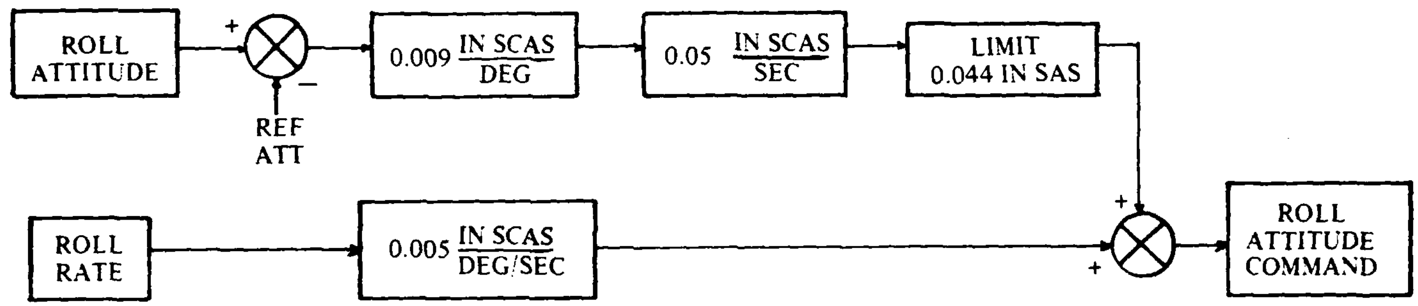


Figure 16. Roll Attitude Hold

HYDRAULIC SYSTEM

General

16. The hydraulic system consists of four hydraulic servoactuators powered simultaneously by two independent 3000-psi hydraulic systems. Each servoactuator simultaneously receives pressure from the primary and utility systems to drive the dual-tandem actuators. This design allows the remaining system to automatically continue powering the servos in the event of a single hydraulic system failure. The two systems (primary and utility) are driven by the accessory gearbox utilizing variable displacement pumps, independent reservoirs and accumulators. The Auxiliary Power Unit (APU) drives all accessories, including the hydraulic pumps, when the aircraft is on the ground and the rotor is not turning. The accessory gearbox is driven by the main transmission during flight and provides for normal operation of both hydraulic systems during autorotation. An emergency hydraulic system is provided to allow emergency operation of the flight controls in the event of a dual system failure.

Primary Hydraulic System

17. The primary hydraulic system (fig 17) consists of a one-pint capacity reservoir, which is pressurized to 30 psi using air from the shaft-driven compressor; an accumulator, which has a nitrogen precharge of 1600 psi, designed to reduce surges in the hydraulic system; and a primary manifold that directs the fluid to the lower side of the four servoactuators. The primary system also provides the hydraulic pressure for operation of the DASE and BUCS functions.

Utility Hydraulic System

18. The utility hydraulic system (fig 18) consists of an air pressurized 1.3 gallon reservoir and a 3000-psi accumulator which drives the APU starting motor. The utility manifold directs fluid to the upper side of the servoactuators, the stores pylon system, tail wheel lock mechanism, area weapon turret drive, and rotor brake. Other manifold functions include an auxiliary isolation check valve which isolates the area weapon turret drive and external stores actuators when either a low pressure or low fluid condition exists; a low pressure sensor isolates the accumulator as an emergency hydraulic source for the servoactuators in the event of a dual hydraulic system failure. The accumulator assembly stores enough fluid for emergency operation of the flight controls through four full strokes of the collective stick and one 180 degrees heading change. The emergency system may be activated by either the pilot's or CPG's emergency switch. An electrically activated emergency shutoff valve is designed to isolate the utility side of the directional servoactuator and the tail wheel lock mechanism when a low fluid condition exists. However, the electrical connections were not installed during EDT-4.

Servoactuators

19. Individual hydraulic servoactuators are provided for longitudinal, lateral, collective, and directional controls. Each servoactuator (fig 19) consists of a ballistically tolerant housing, a single actuator rod with dual frangible pistons, a LAP assembly, BUCS plunger, and various parts for routing of both primary and utility hydraulic fluid. The system is designed to accomodate all flight loads with a failure of either system. However, DASE and BUCS functions would be lost with

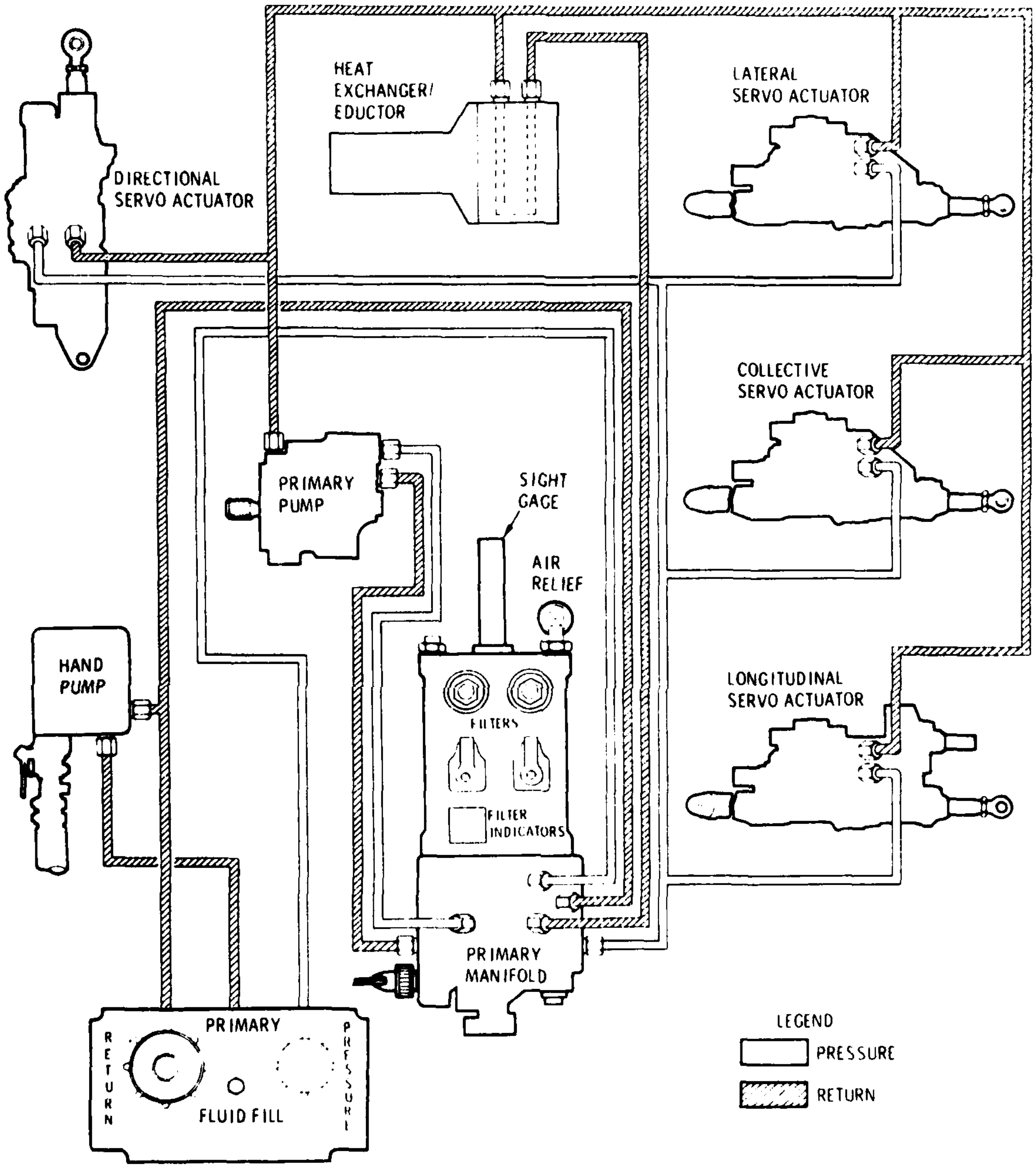


Figure 17. Primary Hydraulic System

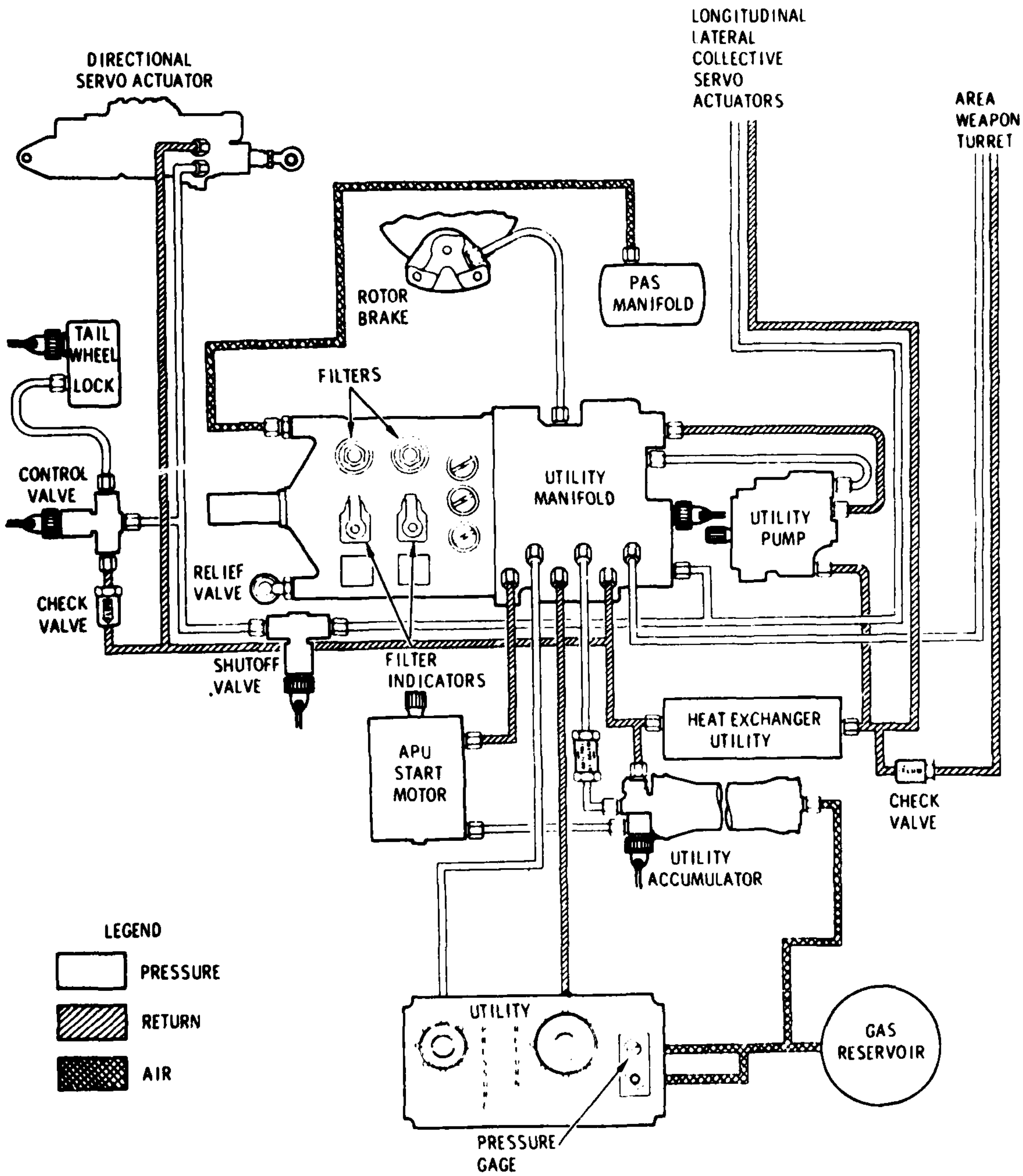


Figure 18. Utility Hydraulic System

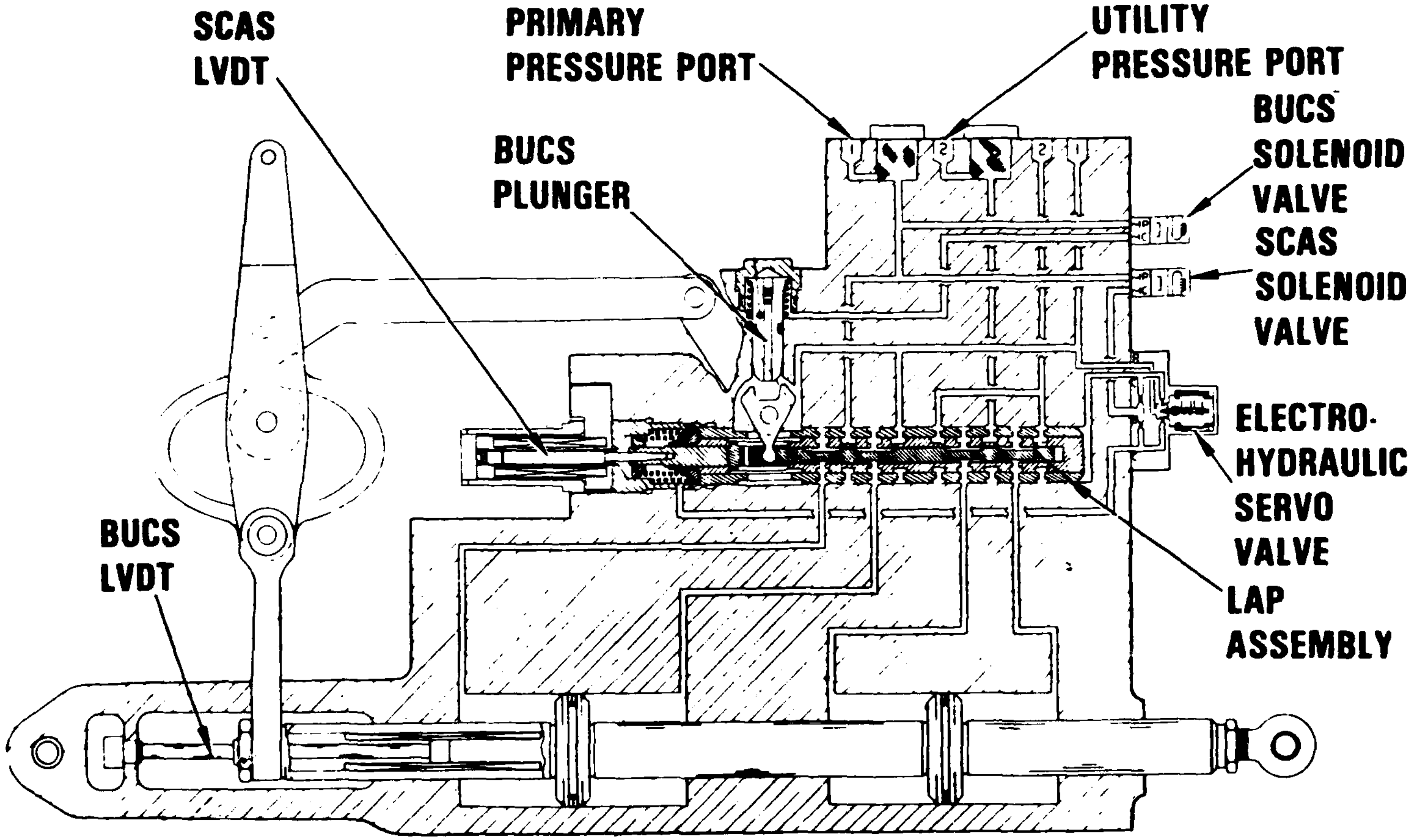


Figure 19. Flight Control Servoactuator

failure of the primary system. The BUCS plunger assemblies were installed during EDT-4, however, electrical connections were omitted.

POWER PLANT

20. The power plant for the YAH-64 helicopter is the General Electric YT700-GE-700R front drive turboshaft engine, rated at 1563 shp (sea level, standard day, uninstalled). The engines are mounted in nacelles on either side of the main transmission. The basic engine consists of four modules: A cold section, a hot section, a power turbine, and an accessory section. Design features of each engine include an axial-centrifugal flow compressor, a through-flow combustor, a two-stage air-cooled high-pressure gas generator turbine, a two-stage uncooled power turbine, and self-contained lubrication and electrical systems. In order to reduce sand and dust erosion, and foreign object damage, an integral particle separator operates when the engine is running. The YT700-GE-700R engine also incoporporates a history recorder which records total engine events. Engines S/N 207-263R and S/N 207-277R were installed in the left and right positions, respectively. Pertinent engine data are shown below:

Model	YT700-GE-700R
Type	Turboshaft
Rated power (intermediate)	1563 shp sea level, standard day, uninstalled
Output speed (at 100 percent N_R)	20,952 rpm
Compressor	5 axial stages, 1 centrifugal stage
Variable geometry	Inlet guide vanes, stages 1 and 2 Stator vanes
Combustion chamber	Single annular chamber with axial flow
Gas generator turbine stages	2
Power turbine stages	2
Direction of rotation (aft looking forward)	Clockwise
Weight (dry)	415 lb
Length	46.5 in.
Maximum diameter	25 in.
Fuel	MIL-T-5624 (JP-4 or JP-5)
Lubricating oil	MIL-L-7808 or MIL-L-23699

Electrical power requirements for history recorder and N_P overspeed protection	40W, 115VAC, 400 Hz
Electrical power requirements for anti-ice valve, filter bypass indication, oil filter bypass indication, and magnetic chip detector	1 amp, 28 VDC

INFRARED (IR) SUPPRESSION SYSTEM

21. The IR suppression system was designed to replace the engine cooling fan used in the Phase 1 aircraft. The IR suppression consists of finned exhaust pipes attached to the engine outlet and bent outboard to mask hot engine parts. The finned pipes radiate heat which is cooled by rotor downwash in hover and turbulent air flow in forward flight. The engine exhaust plume is cooled by mixing it with engine cooling air and bay cooling air (fig 20). The exhaust acts as an eductor, creating air flow over the combustion section of the engine providing engine cooling. Fixed louvers on the top and bottom of the aft cowl and a door on the bottom forward cowling provide convective cooling to the engine during shutdown. The movable bottom door is closed by engine bleed air during engine operation.

FUEL SYSTEM

22. The YAH-64 fuel system has two fuel cells located fore and aft of the ammunition bay. The system includes a fuel boost pump in the aft cell for starting and for high-altitude operation, a fuel transfer pump for transferring fuel between cells, a fuel crossfeed/shutoff valve, and provisions for pressure and gravity fueling and defueling. Additionally, provisions exist for external, wing-mounted fuel tanks. Figure 21 is a schematic of the fuel system. Figure 22 shows the locations and capacities of the two internal fuel cells.

23. By using the tank select switch on his fuel control panel (fig 23), the pilot can select either or both tanks from which the engines will draw fuel. With the tank select switch in the NRML position, the left (No. 1) engine will draw fuel from the forward fuel cell and the right (No. 2) engine will draw from the aft cell (fig 24). When FROM FWD is selected on the tank select switch, the two fuel crossfeed/shutoff valves are positioned so that both engines draw fuel from the forward tank (fig 25). The FROM AFT position allows the engines to draw fuel from the aft tank only (fig 26). The tank select switch is disabled whenever the boost pump is on. When the boost pump is on, the fuel crossfeed/shutoff valves are positioned to allow only fuel from the aft cell to feed both engines (fig 27). The air-driven boost pump operates automatically during engine start and may be activated by the switch on the pilot or CPG fuel control panel (fig 23).

24. The pilot and CPG also have the capability to transfer fuel between tanks using the transfer switch on the fuel control panels (fig 23). Moving the fuel transfer switch out of the OFF position closes the refuel valve and starts the air-driven pump transferring fuel in the selected direction (fig 28).

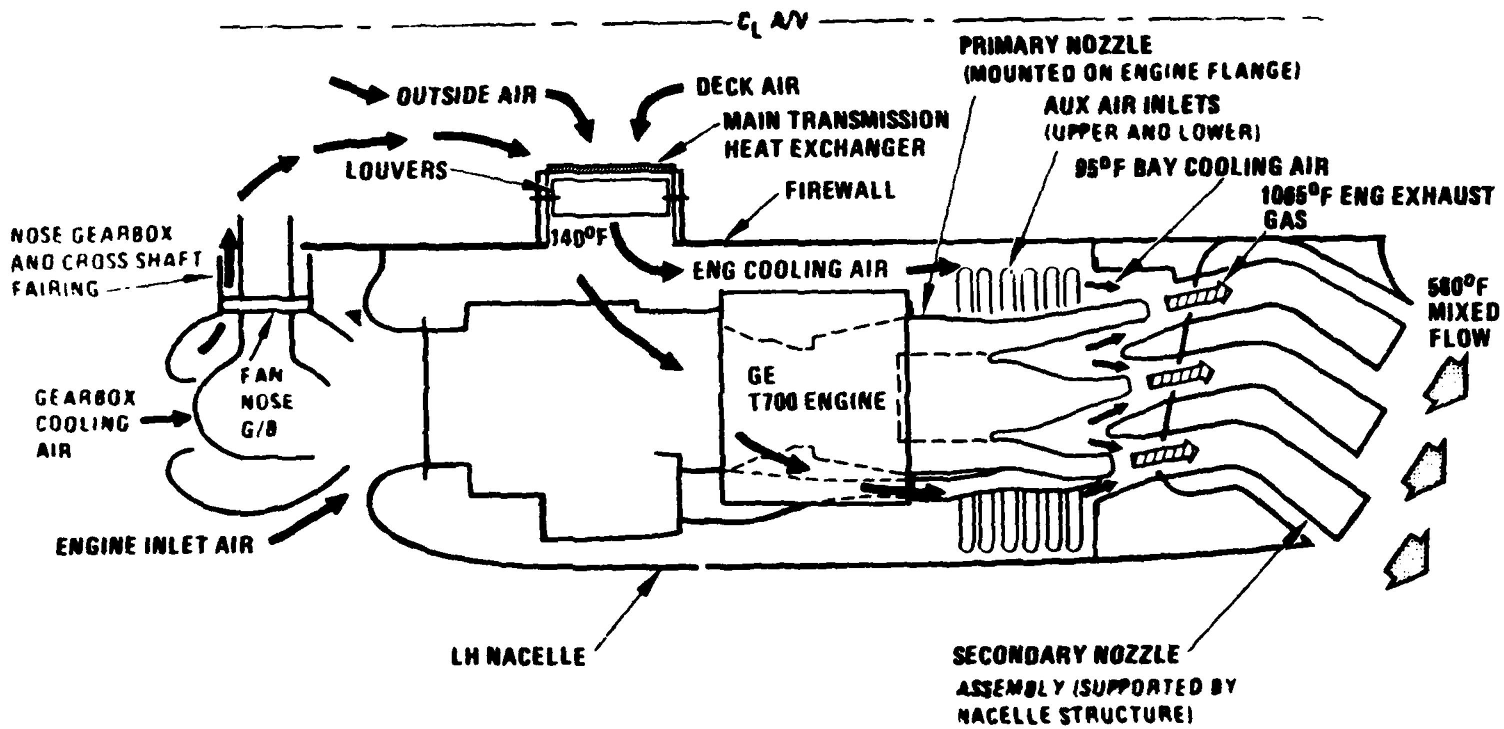


Figure 20. Infrared Suppression System Engine Cooling

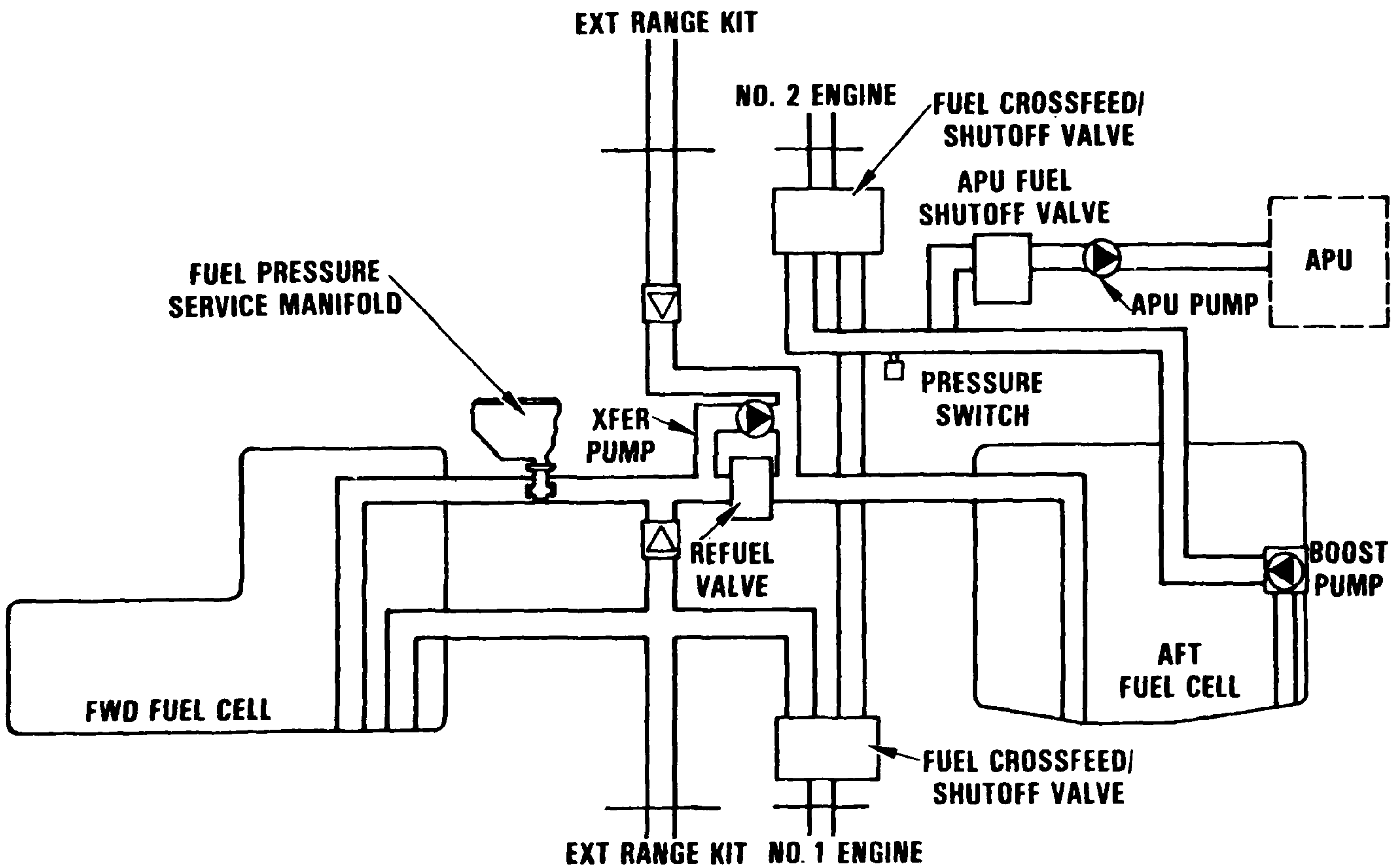


Figure 21. Fuel System Major Components

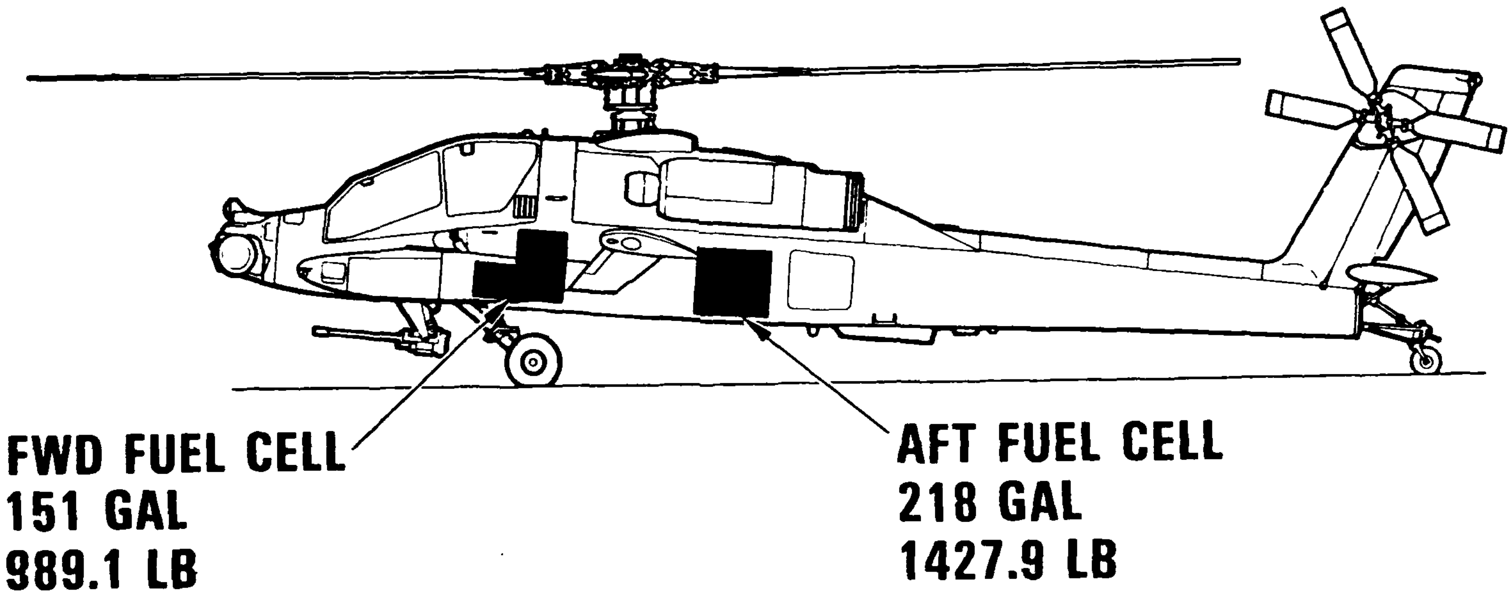


Figure 22. Fuel Cell Locations

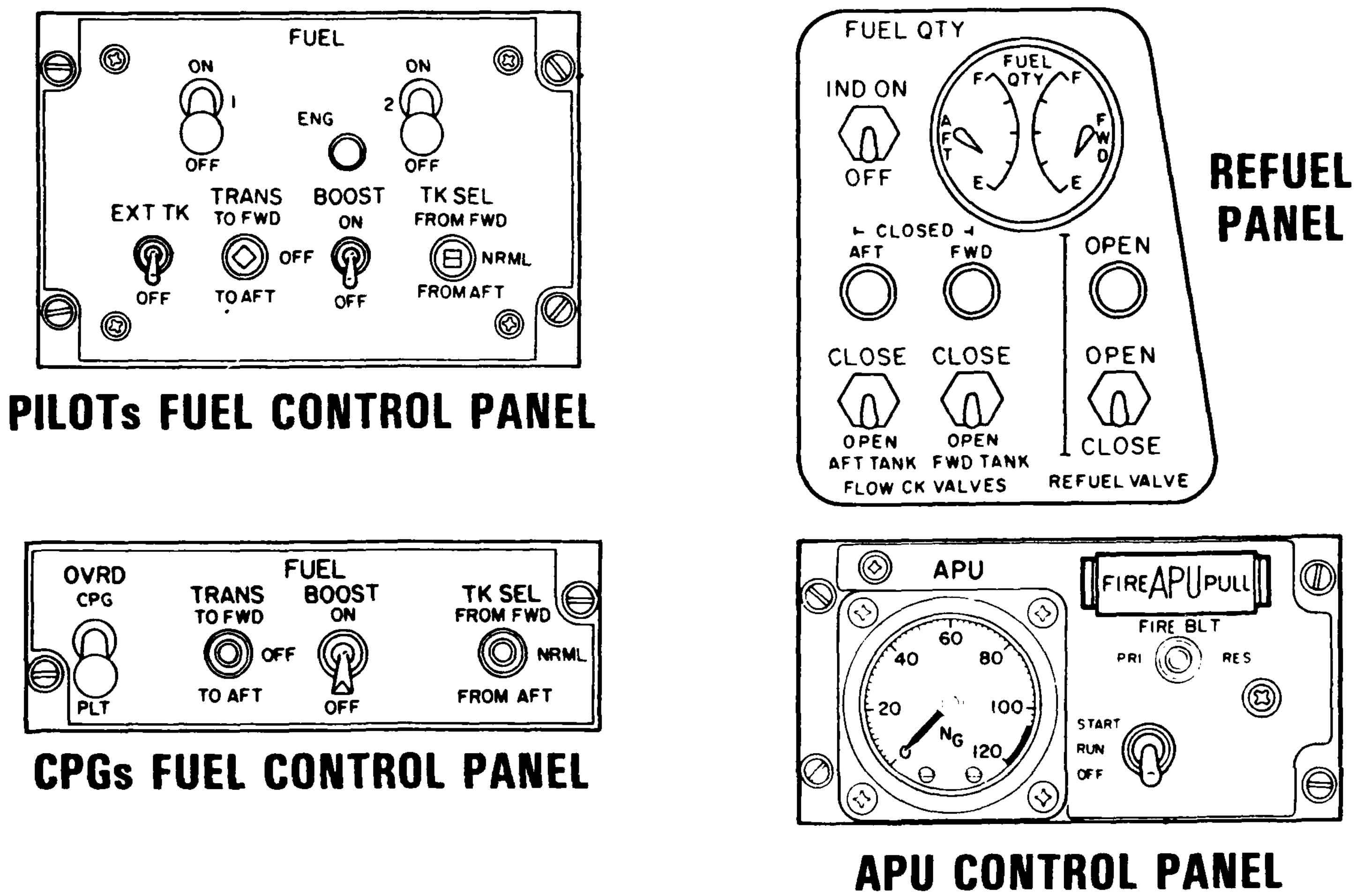


Figure 23. Fuel System Controls

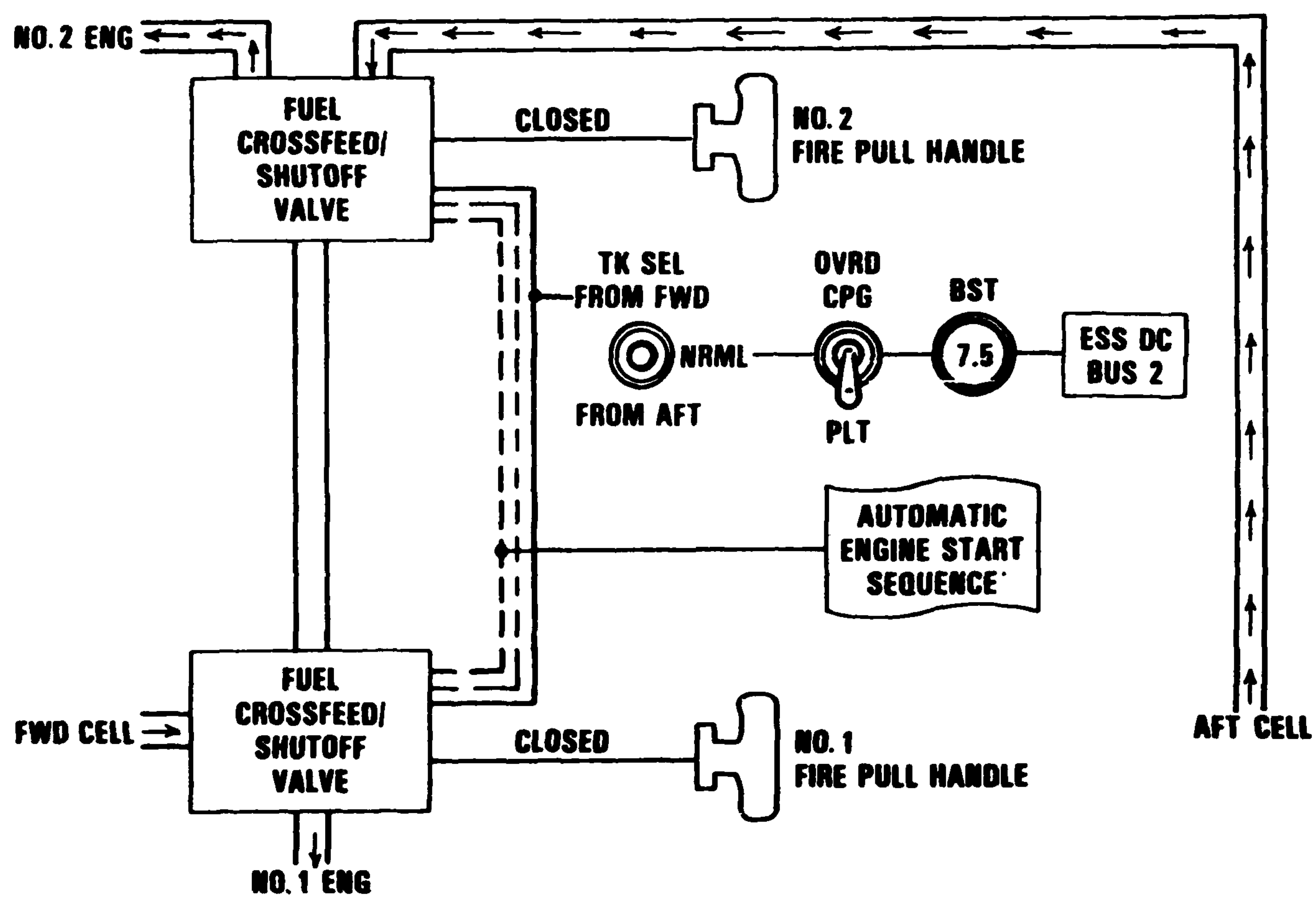


Figure 24. Fuel Crossfeed Operation Normal

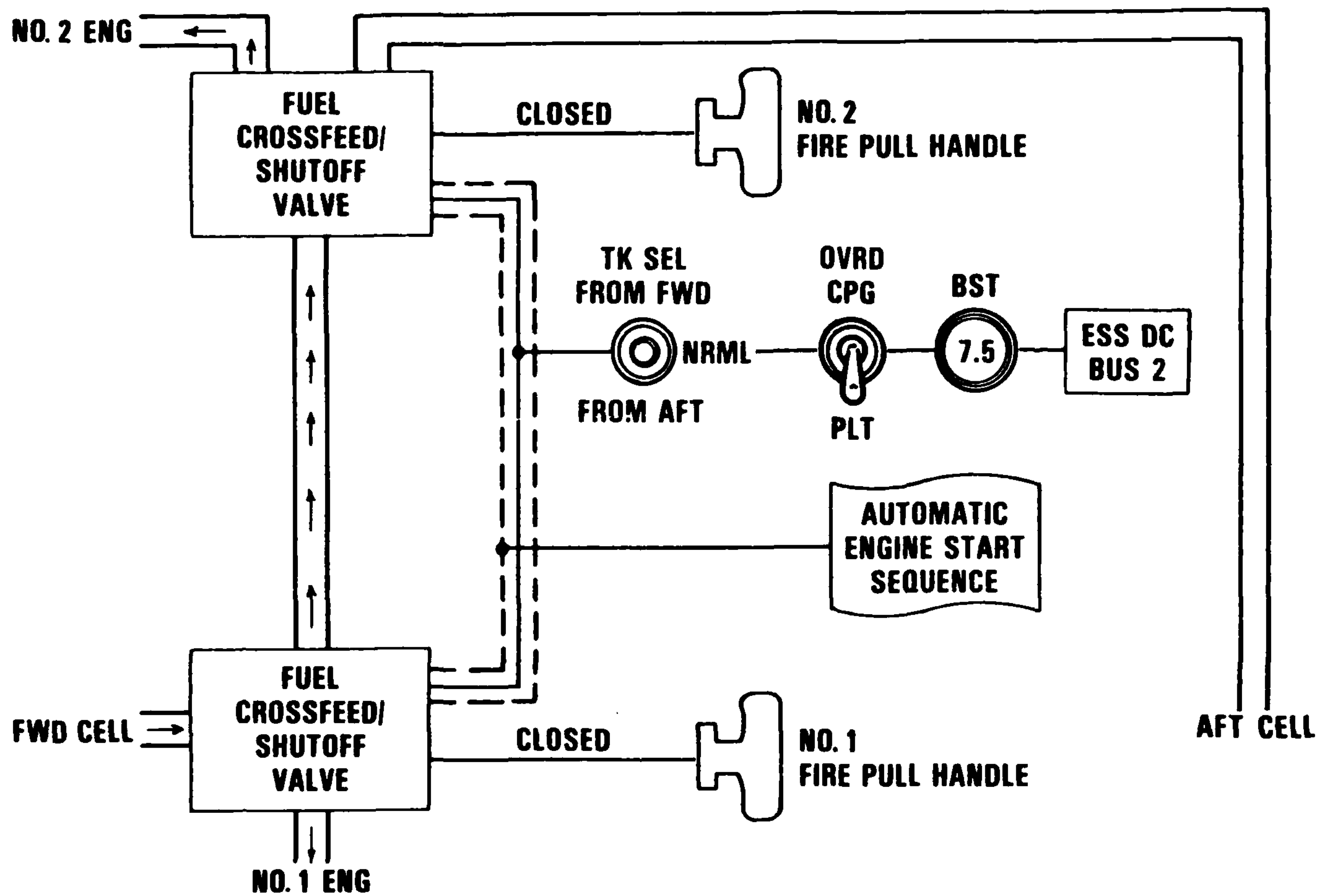


Figure 25. Fuel Crossfeed Operation from Fwd

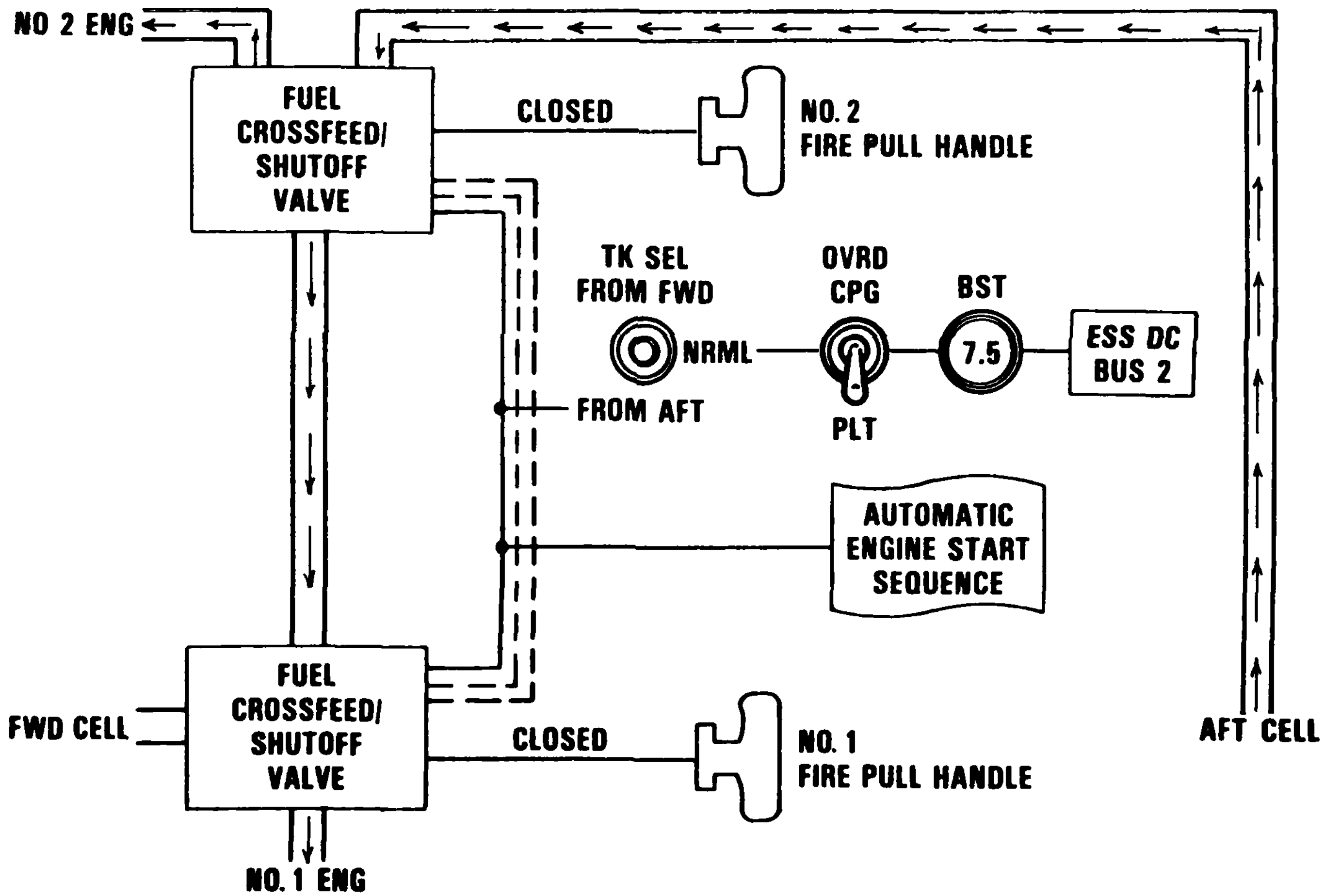


Figure 26. Fuel Crossfeed Operation from Aft

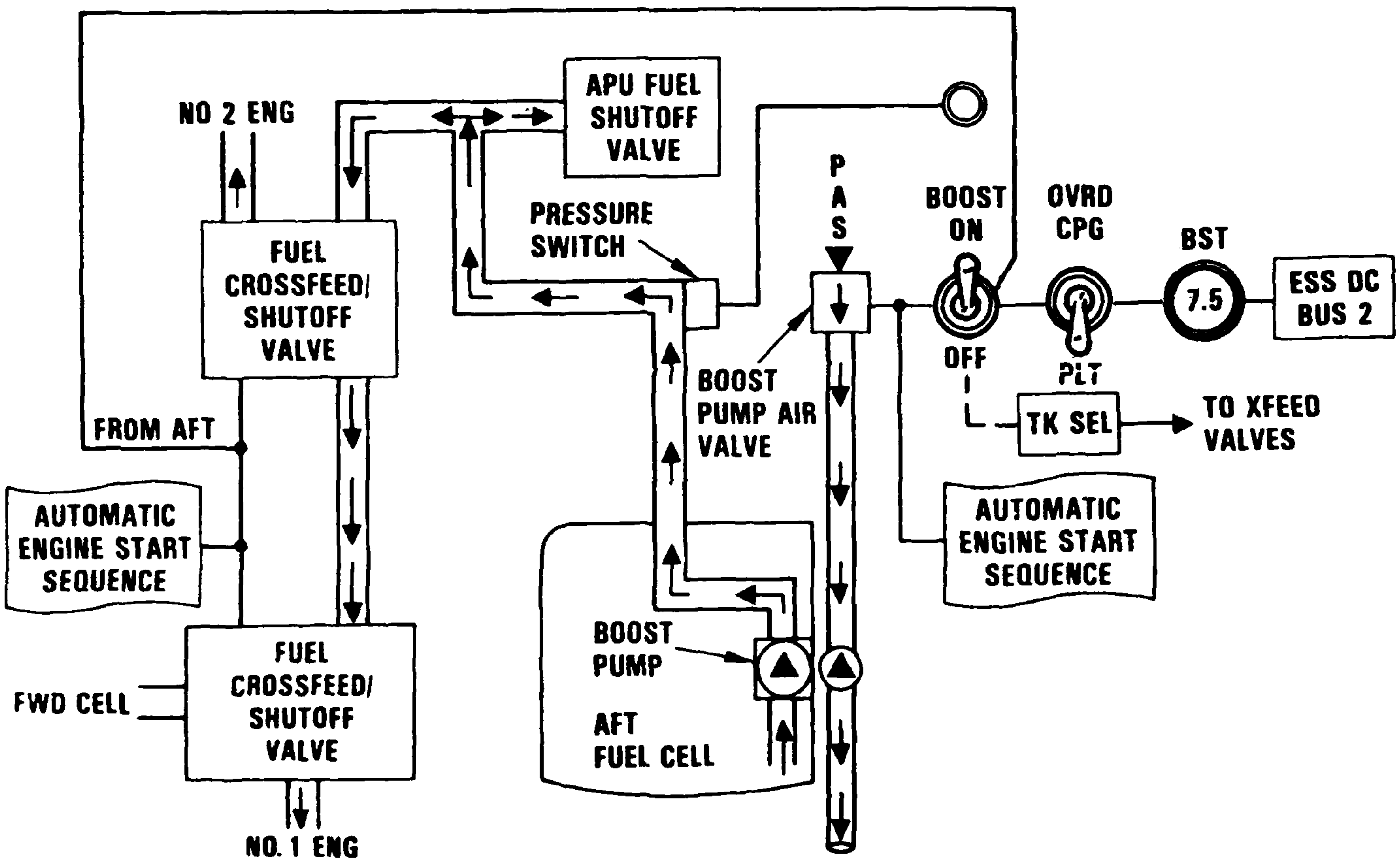


Figure 27. Fuel System Boost Operation

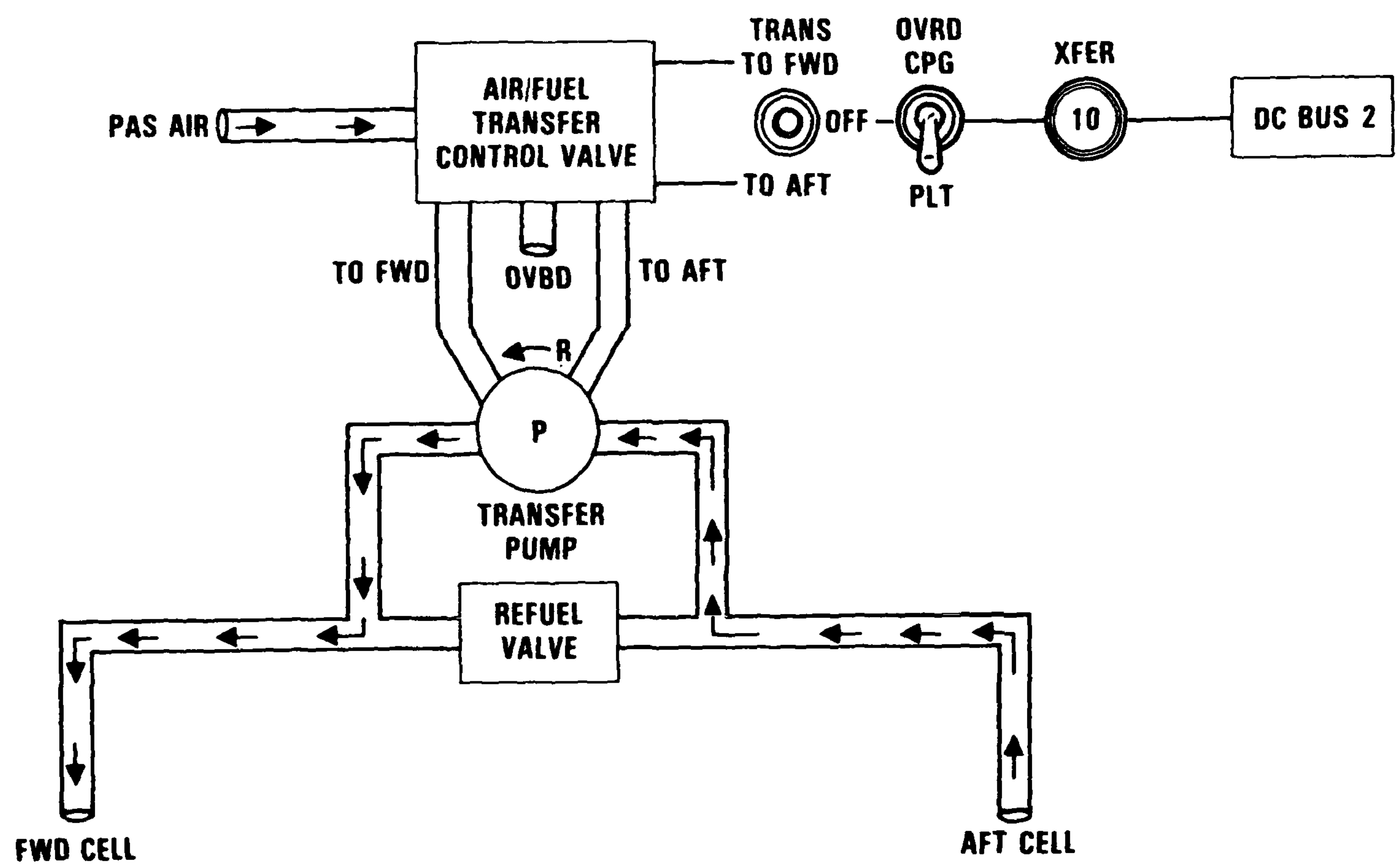


Figure 28. Fuel Transfer Operation

APPENDIX C. INSTRUMENTATION

The airborne data acquisition system was installed, calibrated, and maintained by Hughes Helicopters. The system used pulse code modulation (PCM) encoding, and magnetic tape was used to record parameters on board the aircraft. A boom was mounted on the left side of the aircraft, extending forward of the nose. A pitot-static tube, an angle-of-attack sensor, and an angle-of-sideslip sensor were mounted on the boom. Instrumentation and related special equipment installed in the aircraft and used for this test are shown in the following lists:

Pilot Station (Aft Cockpit)

- Pressure altitude (boom)
- Pressure altitude (ship)
- Airspeed (boom)
- Cable tension
- Airspeed (right-hand ship system)
- Main rotor speed
- Engine torque (both engines)*
- Engine measured gas temperature (both engines)*
- Engine power turbine speed (both engines)*
- Engine gas producer speed (both engines)*
- Angle of sideslip
- Event switch
- Tether cable angles (longitudinal and lateral)
- Longitudinal control position
- Lateral control position
- Directional control position
- Collective control position
- Stabilator angle*

Copilot/Engineer Station

- Airspeed (boom)
- Altitude (boom)
- Main rotor speed
- Engine torque (both engines)*
- Engine measured gas temperature (both engines)*
- Engine gas producer speed (both engines)*
- Total air temperature
- Time code display
- Event switch
- Data system controls

*Standard aircraft instruments

PCM Parameters

- Time code
- Event
- Main rotor speed
- Fuel temperature (both engines)
- Fuel used (both engines)
- Engine fuel flow rate (both engines)
- Engine gas producer speed (both engines)

Engine power turbine speed (both engines)
Airspeed (boom)
Airspeed (ship, right and left)
Altitude (boom)
Altitude (ship)
Total air temperature
Angle of attack
Angle of sideslip
Tether cable tension
Tether cable angle (longitudinal and lateral)
Engine torque (both engines)
Engine measured gas temperature (both engines)
Control positions:
- Longitudinal cyclic
- Lateral cyclic
- Pedal
- Collective
- Stabilator

Aircraft attitudes:
- Pitch
- Roll
- Yaw

Aircraft angular velocities:
- Pitch
- Roll
- Yaw

Vibration accelerometers:
- Pilot station (3 axes) (pilot seat)
- Pilot floor (3 axes)
- Copilot station (3 axes) (copilot station)
- Center of gravity (3 axes)

Stability augmentation system actuator positions:
- Longitudinal
- Lateral
- Directional

Engine power available spindle (both engines)
Air data system:
- ADS Vx
- ADS Vy
- ADS Vz
- ADS Hp
- ADS OAT

APPENDIX D. TEST TECHNIQUES AND DATA ANALYSIS METHODS

DATA ANALYSIS METHODS

1. The helicopter performance test data were generalized by use of nondimensional coefficients and were such that the effects of compressibility and blade stall were not separated and defined. The following nondimensional coefficients were used to generalize the hover and level flight test results obtained during this flight test program.

a. Coefficient of power (C_P):

$$C_P = \frac{SHP(550)}{\rho A(\Omega R)^3} \tag{1}$$

b. Coefficient of thrust (C_T):

$$C_T = \frac{\text{Thrust}}{\rho A(\Omega R)^2} \tag{2}$$

c. Advance ratio (μ):

$$\mu = \frac{1.6878\ V_T}{\Omega R} \tag{3}$$

d. Advancing tip Mach number (M_{tip}):

$$M_{tip} = \frac{1.6878\ V_T + (\Omega R)}{a} \tag{4}$$

Where:

SHP = Engine output shaft horsepower (both engines)
550 = Conversion factor (ft-lb/sec)/shp
ρ = Air density (slug/ft^3)
A = Main rotor disc area (ft^2)
Ω = Main rotor angular velocity (radian/sec)
R = Main rotor radius (ft)
Thrust = Gross weight (lb) during free flight in which there is no acceleration or velocity component in the vertical direction. Tether load must be added in the case of tethered hover.
1.6878 = Conversion factor (ft/sec)/kt
V_T = True airspeed (kt)
a = Speed of sound (ft/sec) = $1116.45\sqrt{\theta}$

SHAFT HORSEPOWER REQUIRED

2. Engine output shaft torque was determined by the use of the engine torquemeter. The torquemeter was calibrated in a test cell by the engine manufacturer. The outputs from the engine torquemeters were recorded on the onboard data recording system. The output shp was determined from the engine

output shaft torque and rotational speed by the following equation:

$$SHP = \frac{2\pi \times N_P \times Q}{33{,}000} \qquad (5)$$

Where:

N_P = Engine output shaft rotational speed (rpm)
Q = Engine output shaft torque (ft-lb)
33,000 = Conversion factor (ft-lb/min)/shp

LEVEL FLIGHT PERFORMANCE

3. Level flight performance data were reduced to nondimensional form using equations 1, 2, and 3. Each speed-power was flown at a predetermined C_T with rotor speed held constant. To maintain the ratio of gross weight to air density ratio (W/σ) constant, altitude was increased as fuel was consumed.

4. Test-day (measured) level flight power was corrected to standard-day conditions (average for the flight) by assuming that the test-day dimensionless parameters C_{P_t}, C_{T_t}, and μ_t are identical to C_{P_s}, C_{T_s}, and μ_s, respectively.

From equation 1, the following relationship can be derived:

$$SHP_s = SHP_t \left(\frac{\rho_s}{\rho^t}\right) \qquad (6)$$

Where:

t = test day
s = standard day

5. Test specific range was calculated using level flight performance curves and the measured fuel flow.

$$SR = \frac{V_T}{W_f} \qquad (7)$$

Where:

SR = Specific range (Nautical air miles per pound of fuel)
V_T = True airspeed (kt)
W_f = Fuel flow (lb/hr)

HANDLING QUALITIES

6. Stability and control data were collected and evaluated using standard test methods as described in reference 10, appendix A. Definitions of deficiencies and

shortcomings used during this test are shown below.

Deficiency - A defect or malfunction discovered during the life cycle of an item of equipment that constitutes a safety hazard to personnel; will result in serious damage to the equipment if operation is continued; or indicates improper design or other cause of failure of an item or part, which seriously impairs the equipment's operational capability.

Shortcoming - An imperfection or malfunction occurring during the life cycle of equipment which must be reported and which should be corrected to increase efficiency and to render the equipment completely serviceable. It will not cause an immediate breakdown, jeopardize safe operation, or materially reduce the useability of the material or end product.

VIBRATIONS

7. The PCM vibration data were reduced by means of a fast Fourier transform from the analog flight tape. Vibration levels, representing peak amplitudes, were extracted from this analysis at selected harmonics of the main rotor frequency. The Vibration Rating Scale, presented in figure 1, was used to augment crew comments on aircraft vibration levels.

AIRSPEED CALIBRATION

8. The boom pitot-static system and both ships' systems were calibrated by using the pace aircraft method to determine the airspeed position error. Calibrated airspeed (V_{cal}) was obtained by correcting indicated airspeed (V_i) using instrument (ΔV_{ic}) and position (ΔV_{pc}) error corrections.

$$V_{cal} = V_i + \Delta V_{ic} + \Delta V_{pc} \tag{8}$$

9. True airspeed (V_t) was calculated from the calibrated airspeed and density ratio.

$$V_t = \frac{V_c}{\sqrt{\sigma}} \tag{9}$$

Where:

σ = Density ratio ($\frac{\rho}{\rho_0}$) where ρ_0 is the density at sea level on a standard day.

10. The airspeed from the boom system was used for all data reduction. The calibration of this system is presented as figure 2.

WEIGHT AND BALANCE

11. Prior to testing, the aircraft gross weight and cg were determined by using calibrated scales. The aircraft was weighed with no fuel in the 8-HELLFIRE

DEGREE OF VIBRATION	DESCRIPTION[1]	PILOT RATING
No vibration		0
Slight	Not apparent to experienced aircrew fully occupied by their tasks, but noticeable if their attention is directed to it or if not otherwise occupied.	1 2 3
Moderate	Experienced aircrew are aware of the vibration but it does not affect their work, at least over a short period.	4 5 6
Severe	Vibration is immediately apparent to experienced aircrew even when fully occupied. Performance of primary task is affected or tasks can only be done with difficulty.	7 8 9
Intolerable	Sole preoccupation of aircrew is to reduce vibration level.	10

[1] Based upon the Subjective Vibration Assessment Scale developed by the Aeroplane and Armament Experimental Establishment, Boscombe Down, England.

Figure 1. Vibration Rating Scale

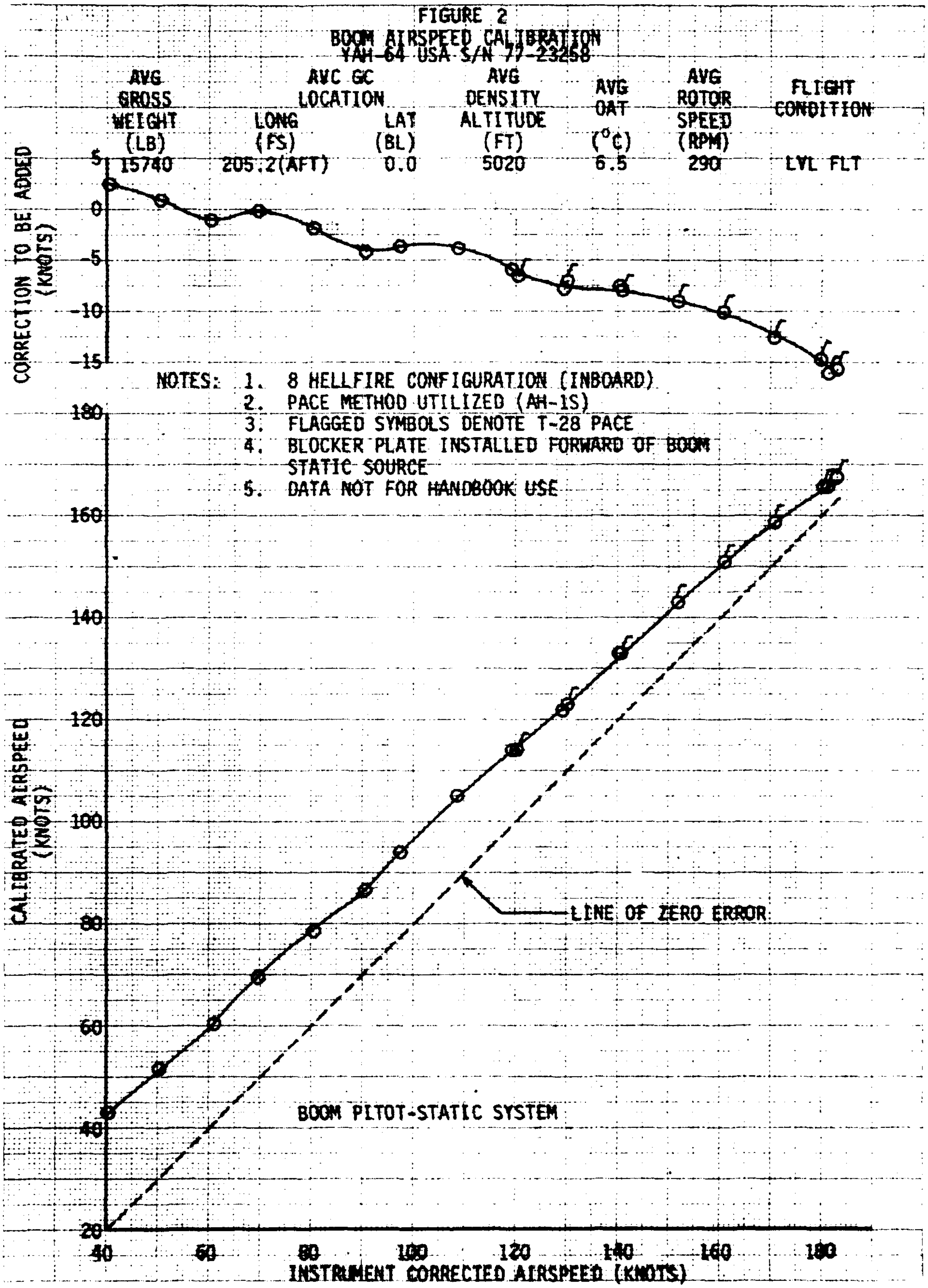
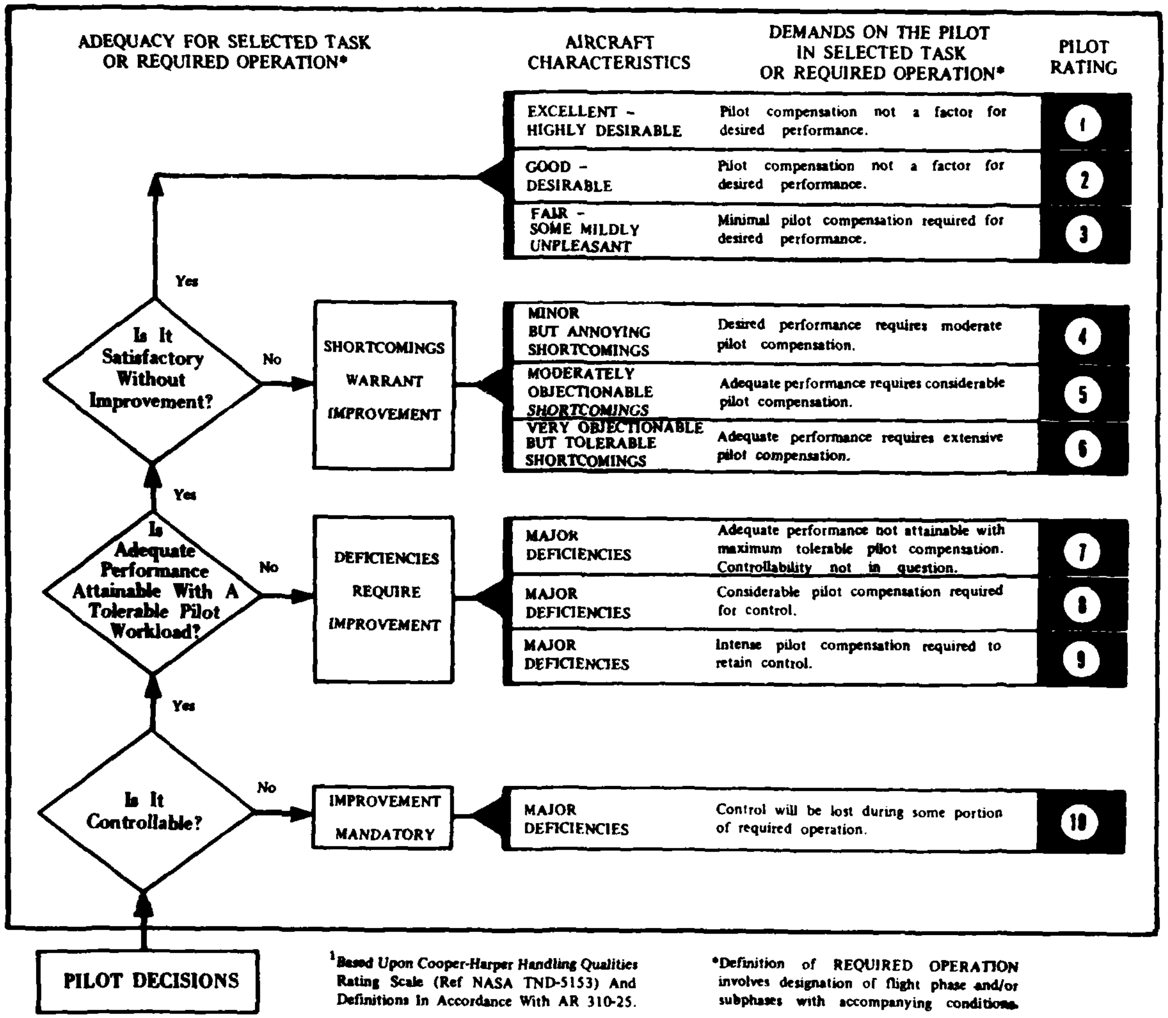


FIGURE 2
BOOM AIRSPEED CALIBRATION
YAH-64 USA S/N 77-23258
AVG GROSS WEIGHT (LB) 15740
AVG GC LOCATION LONG (FS) 205.2(AFT) LAT (BL) 0.0
AVG DENSITY ALTITUDE (FT) 5020
AVG OAT (°C) 6.5
AVG ROTOR SPEED (RPM) 290
FLIGHT CONDITION LVL FLT
CORRECTION TO BE ADDED (KNOTS)
NOTES: 1. 8 HELLFIRE CONFIGURATION (INBOARD)
2. PACE METHOD UTILIZED (AH-1S)
3. FLAGGED SYMBOLS DENOTE T-28 PACE
4. BLOCKER PLATE INSTALLED FORWARD OF BOOM STATIC SOURCE
5. DATA NOT FOR HANDBOOK USE
CALIBRATED AIRSPEED (KNOTS)
LINE OF ZERO ERROR
BOOM PITOT-STATIC SYSTEM
INSTRUMENT CORRECTED AIRSPEED (KNOTS)



[1]Based Upon Cooper-Harper Handling Qualities Rating Scale (Ref NASA TND-5153) And Definitions In Accordance With AR 310-25.

*Definition of REQUIRED OPERATION involves designation of flight phase and/or subphases with accompanying conditions.

Figure 3. Handling Qualities Rating Scale

configuration, with instrumentation on board. The aircraft weight was 13,176 pounds with a longitudinal cg location at FS 208.3.

HANDLING QUALITIES RATING SCALE

12. The Handling Qualities Rating Scale (HQRS) presented in figure 3 was used to augment pilot comments relative to handling qualities and work load.

APPENDIX E. TEST DATA

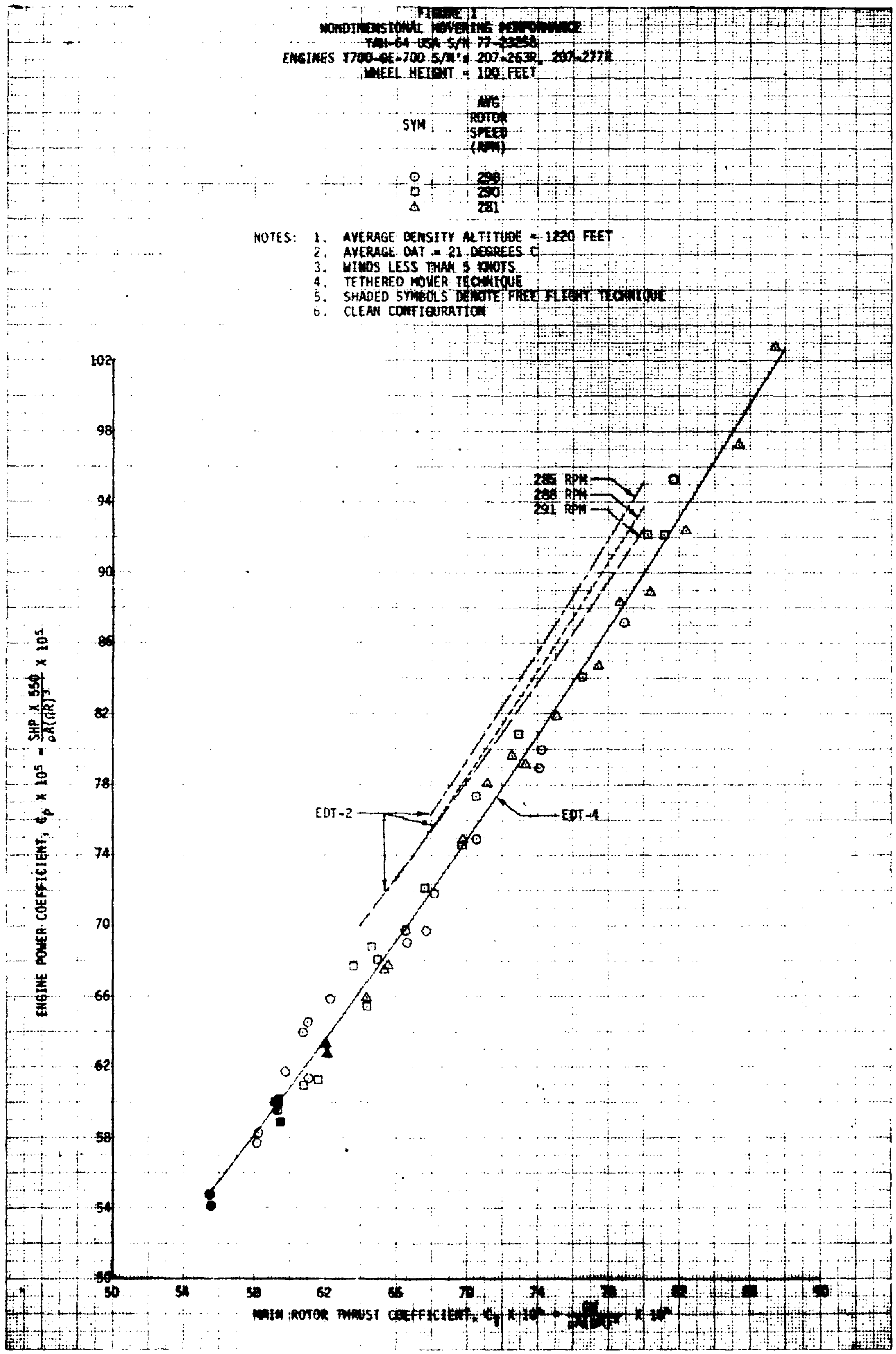


FIGURE 1
NONDIMENSIONAL HOVERING PERFORMANCE
YAH-64 USA S/N 77-23258
ENGINES T700-GE-700 S/N's 207-263R, 207-277R
WHEEL HEIGHT = 100 FEET
SYM
AVG ROTOR SPEED (RPM)
298
290
281
NOTES: 1. AVERAGE DENSITY ALTITUDE = 1220 FEET
2. AVERAGE OAT = 21 DEGREES C
3. WINDS LESS THAN 5 KNOTS
4. TETHERED HOVER TECHNIQUE
5. SHADED SYMBOLS DENOTE FREE FLIGHT TECHNIQUE
6. CLEAN CONFIGURATION
285 RPM
288 RPM
291 RPM
EDT-2
EDT-4
ENGINE POWER COEFFICIENT, $C_P \times 10^5 = \frac{SHP \times 550}{\rho A(\Omega R)^3} \times 10^5$
MAIN ROTOR THRUST COEFFICIENT, $C_T \times 10^4$
102
98
94
90
86
82
78
74
70
66
62
58
54
50
50
54
58
62
66
70
74
78
82
86
90

FIGURE 2
NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE
YAH-64 USA S/N 77-23258

NOTES:
1. AVG LONGITUDINAL CG LOCATION FS 202.0 (FWD)
2. ROTOR SPEED = 290 RPM
3. 8-HELLFIRE CONFIGURATION
4. CURVES DERIVED FROM FIGS 6 THROUGH 9
5. ZERO SIDESLIP

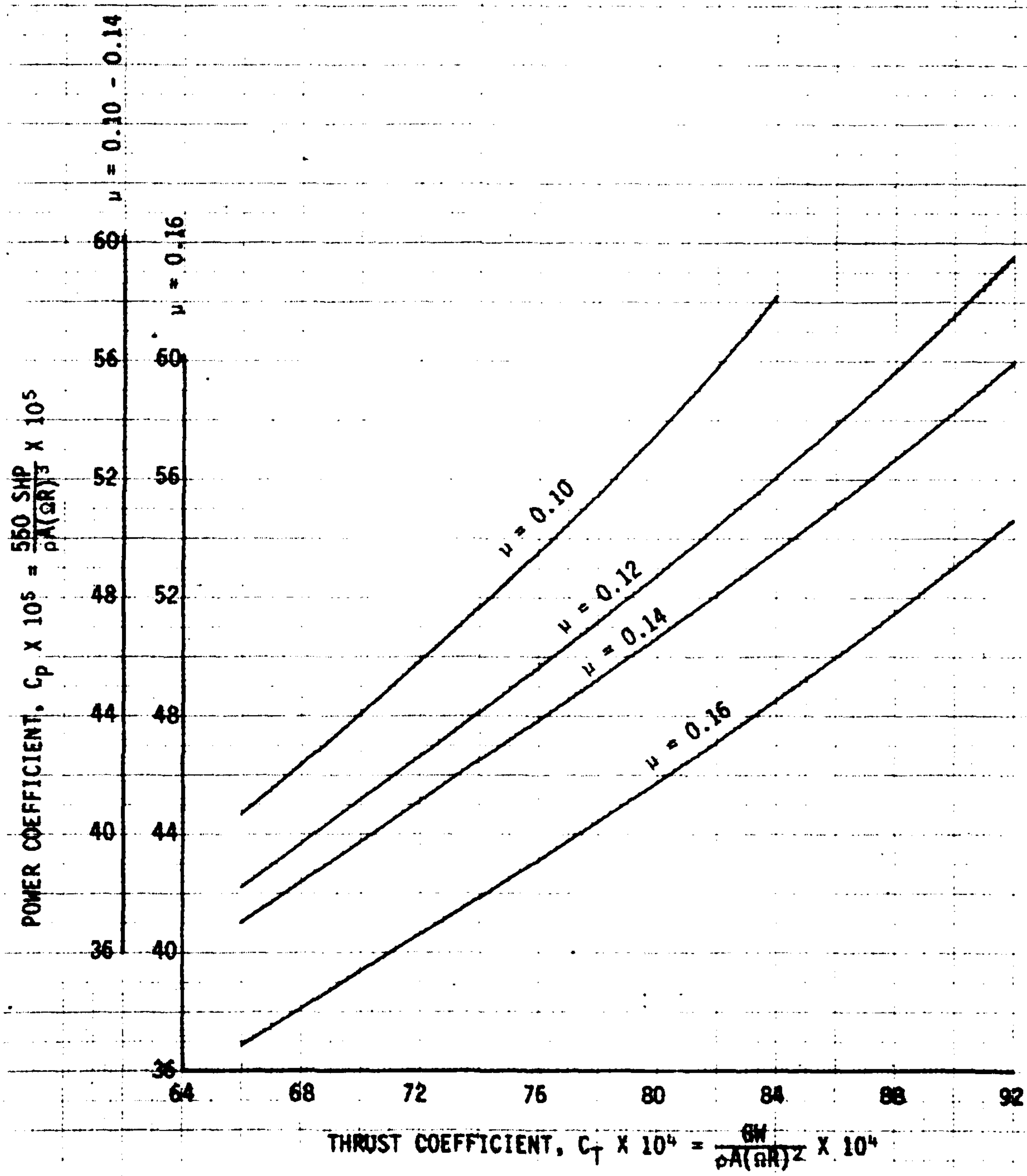


FIGURE 3
NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE
YAH-64 USA S/N 77-23258

NOTES:
1. AVG LONGITUDINAL CG LOCATION FS 202.0 (FWD)
2. ROTOR SPEED = 290 RPM
3. 8-HELLFIRE CONFIGURATION
4. CURVES DERIVED FROM FIGS 6 THROUGH 9
5. ZERO SIDESLIP

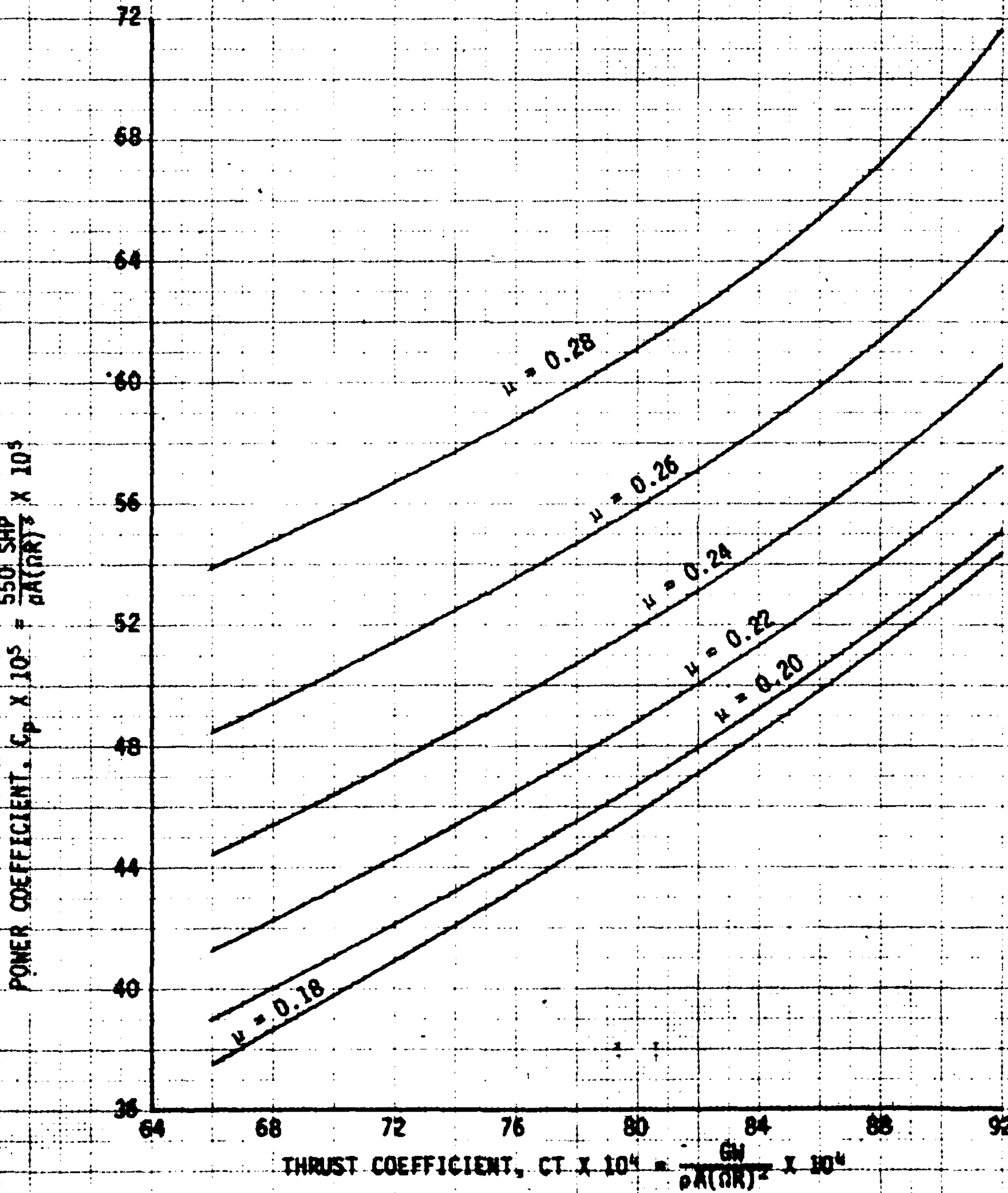


FIGURE 4
NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE
YAH-64 USA S/N 77-23258

NOTES:
1. AVG LONGITUDINAL CG LOCATION FS 202.0 (FWD)
2. ROTOR SPEED = 290 RPM
3. 8-HELLFIRE CONFIGURATION
4. CURVES DERIVED FROM FIGS 6 THROUGH 9
5. ZERO SIDESLIP

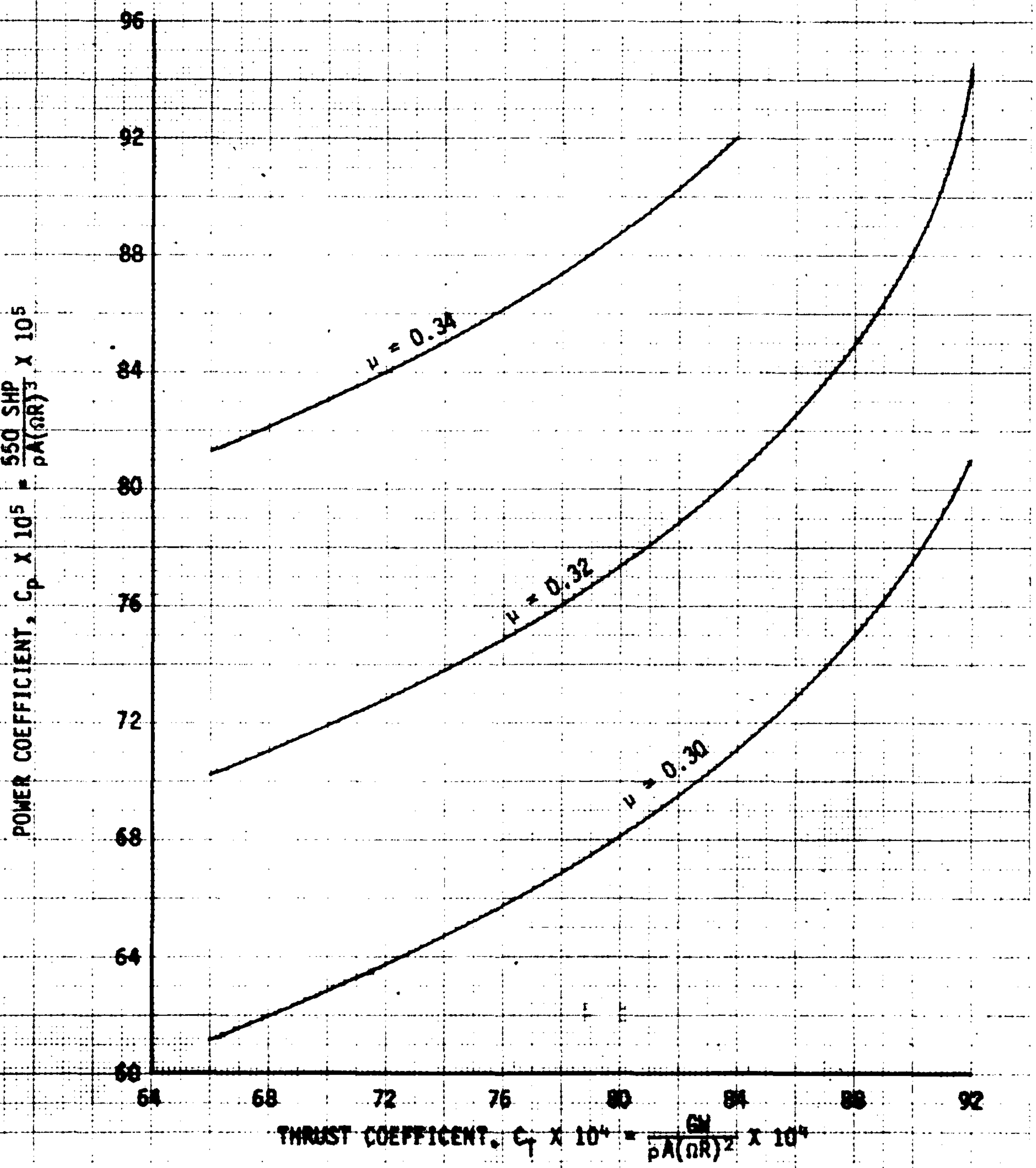


FIGURE 5
LEVEL FLIGHT PERFORMANCE
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG LONG CG LOCATION (FS)	AVG PRESSURE ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG C_T	CONFIG
14200	202 (FWD)	4000	35	289	0.00775	8-HELLFIRE

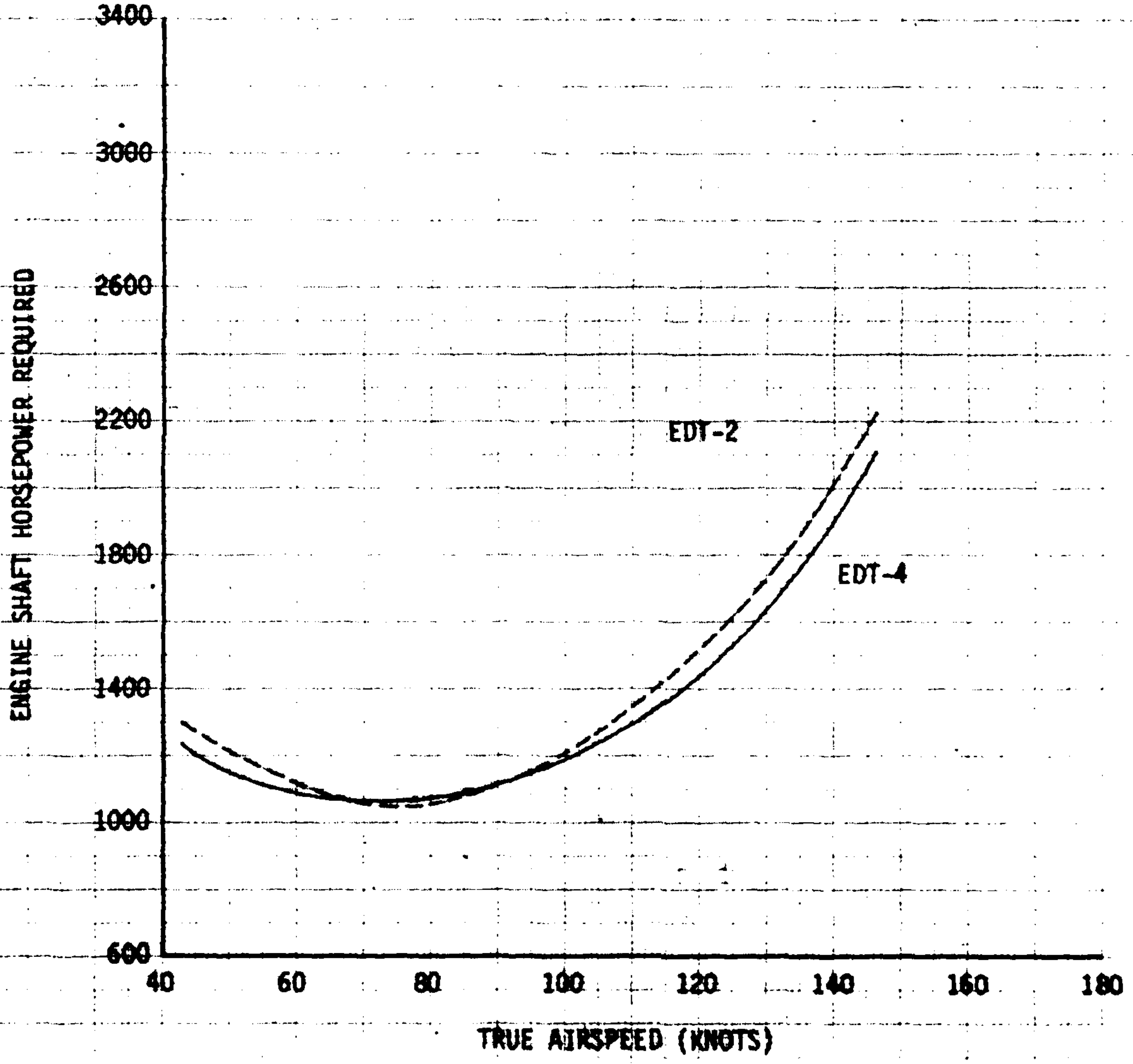


FIGURE 6
LEVEL FLIGHT PERFORMANCE
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT ($^{\circ}C$)	AVG ROTOR SPEED (RPM)	AVG C_{T_4} X10	CONFIGURATION
14600	202.0 (FWD)	0.0	1360	17.5	290	66.51	8-HELLFIRE

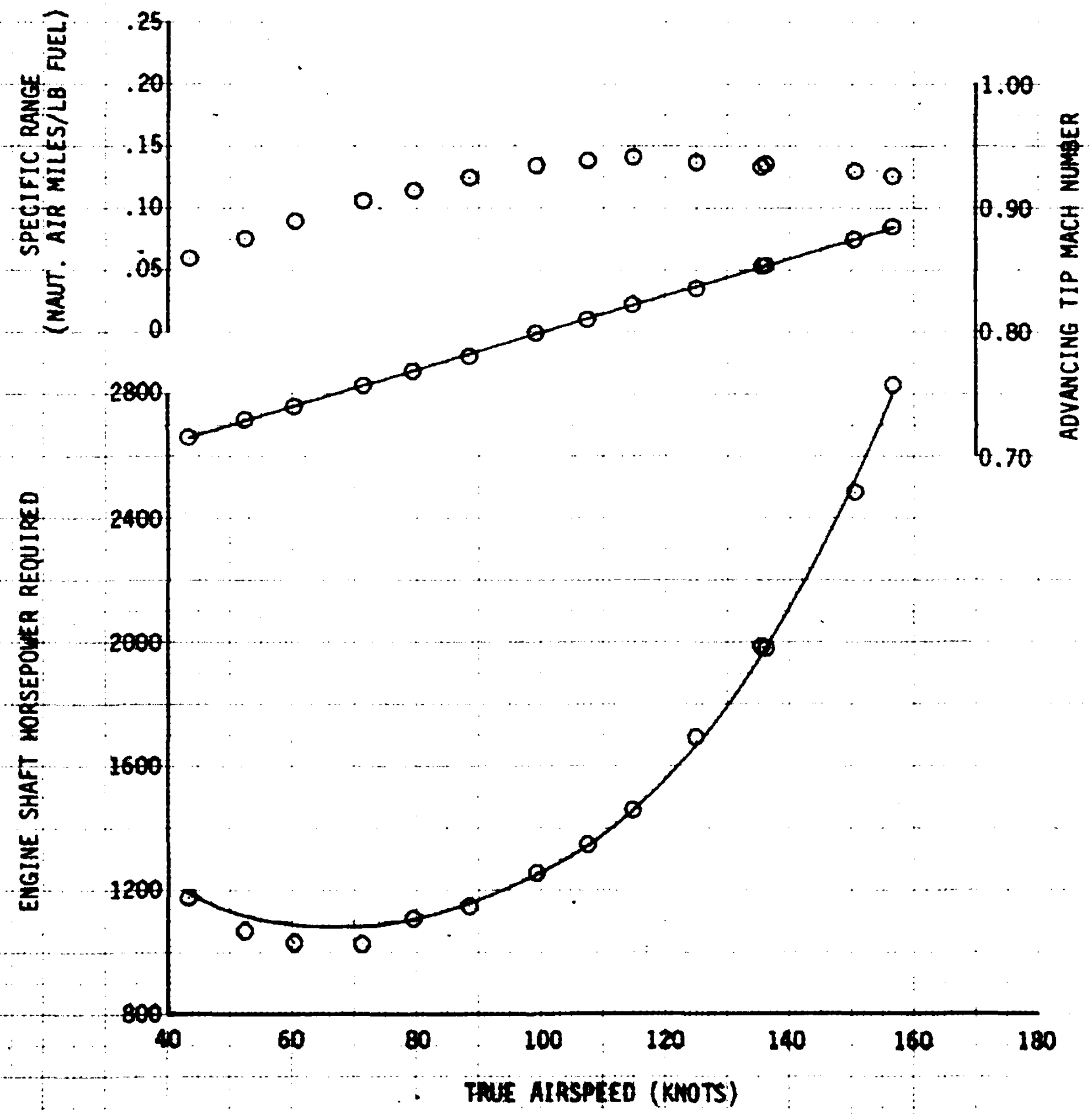


FIGURE 7
LEVEL FLIGHT PERFORMANCE
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG C_{T_4} X10	CONFIGURATION
14860	202.0 (FWD)	0.0	5420	10.0	290	76.44	8-HELLFIRE

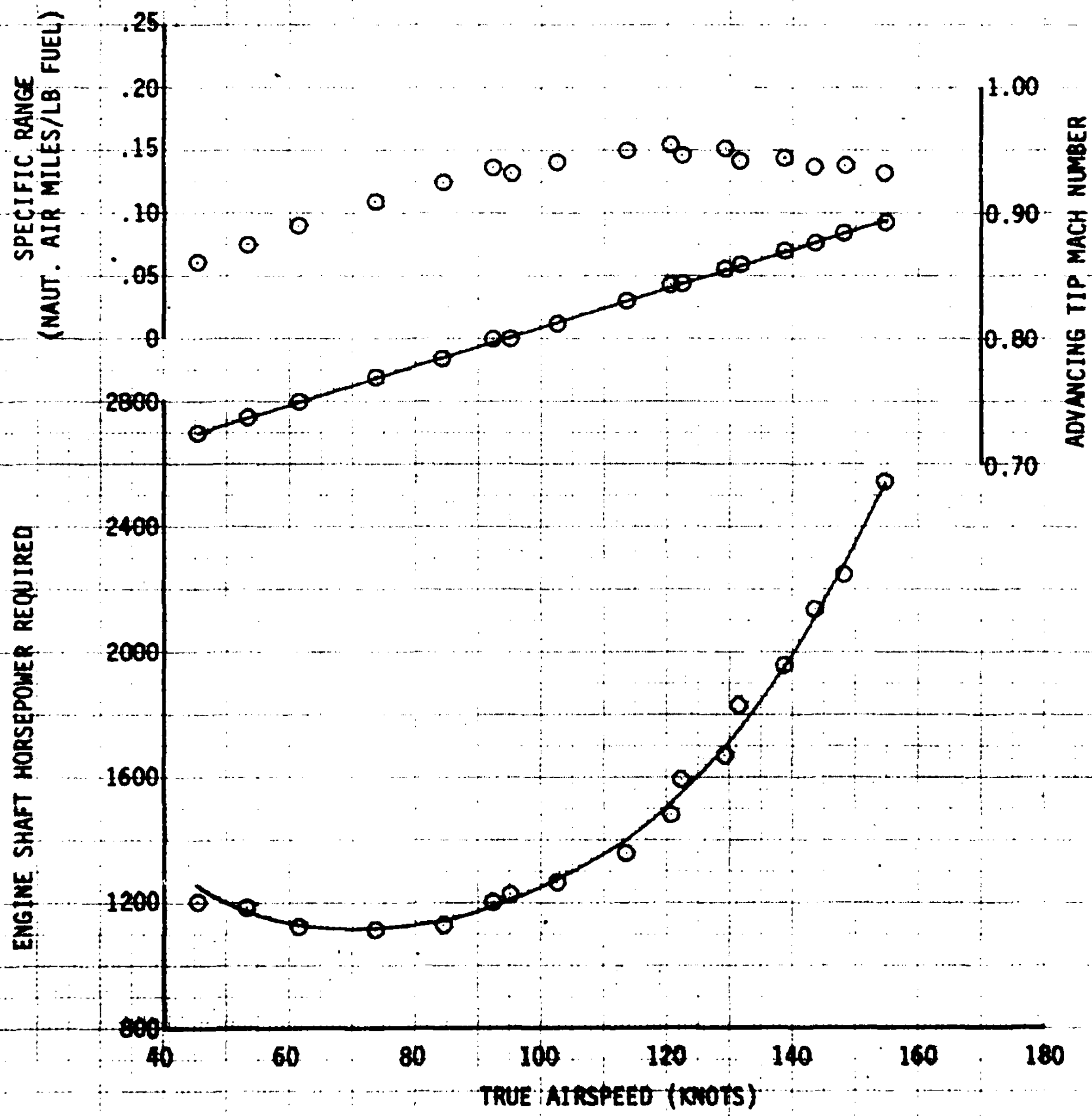


FIGURE 8
LEVEL FLIGHT PERFORMANCE
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG C_{T_4} X10	CONFIGURATION
15740	202.0 (FWD)	0.0	6380	8.0	290	83.37	8-HELLFIRE

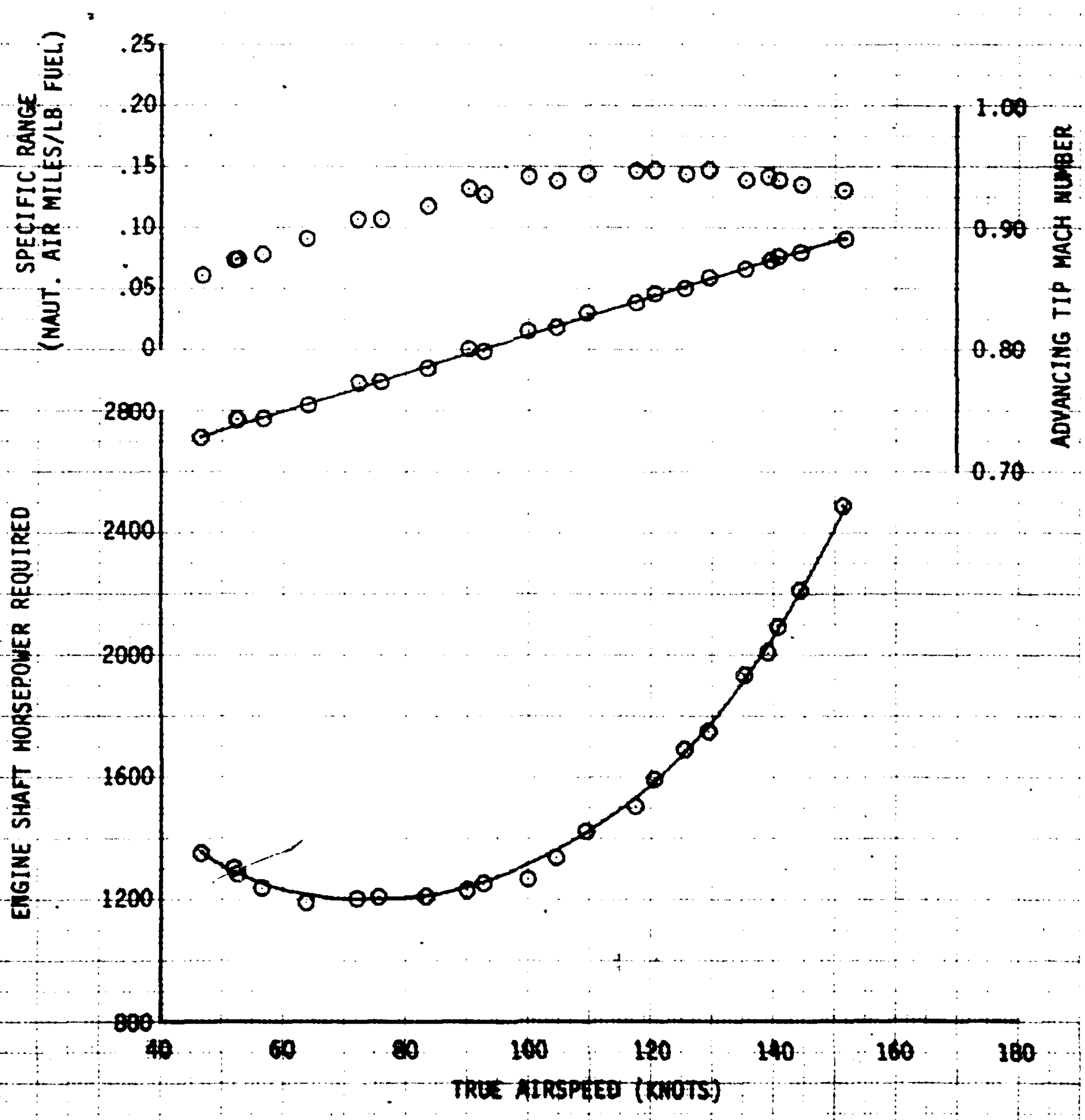


FIGURE 9
LEVEL FLIGHT PERFORMANCE
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG C_{T_4} X10	CONFIGURATION
15640	202.2 (FWD)	0.0	9720	0.5	290	91.87	8-HELLFIRE

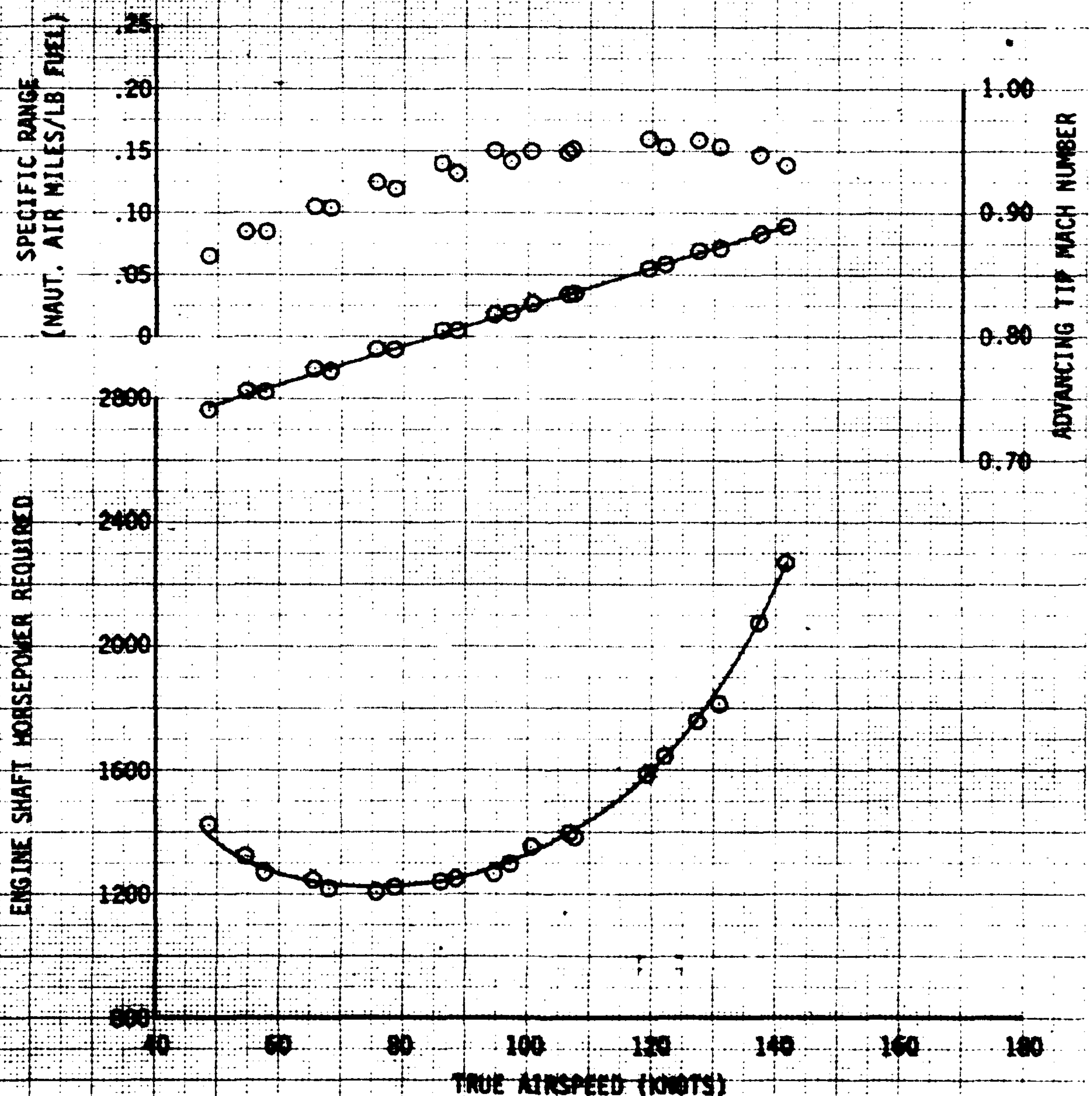


FIGURE 10
LIMITS OF CYCLIC CONTROL TRAVEL
YAH-64 USA S/N 77-23258

NOTES:
1. ROTORS STATIC.
2. HYDRAULIC AND ELECTRICAL POWER PROVIDED BY GROUND POWER UNITS.
3. CONTROL POSITION MEASURED AT CENTER OF GRIP.
4. COLLECTIVE CONTROL FULL DOWN.
5. NO CYCLIC CONTROL PATTERN CHANGE WITH CHANGE OF COLLECTIVE CONTROL POSITION.

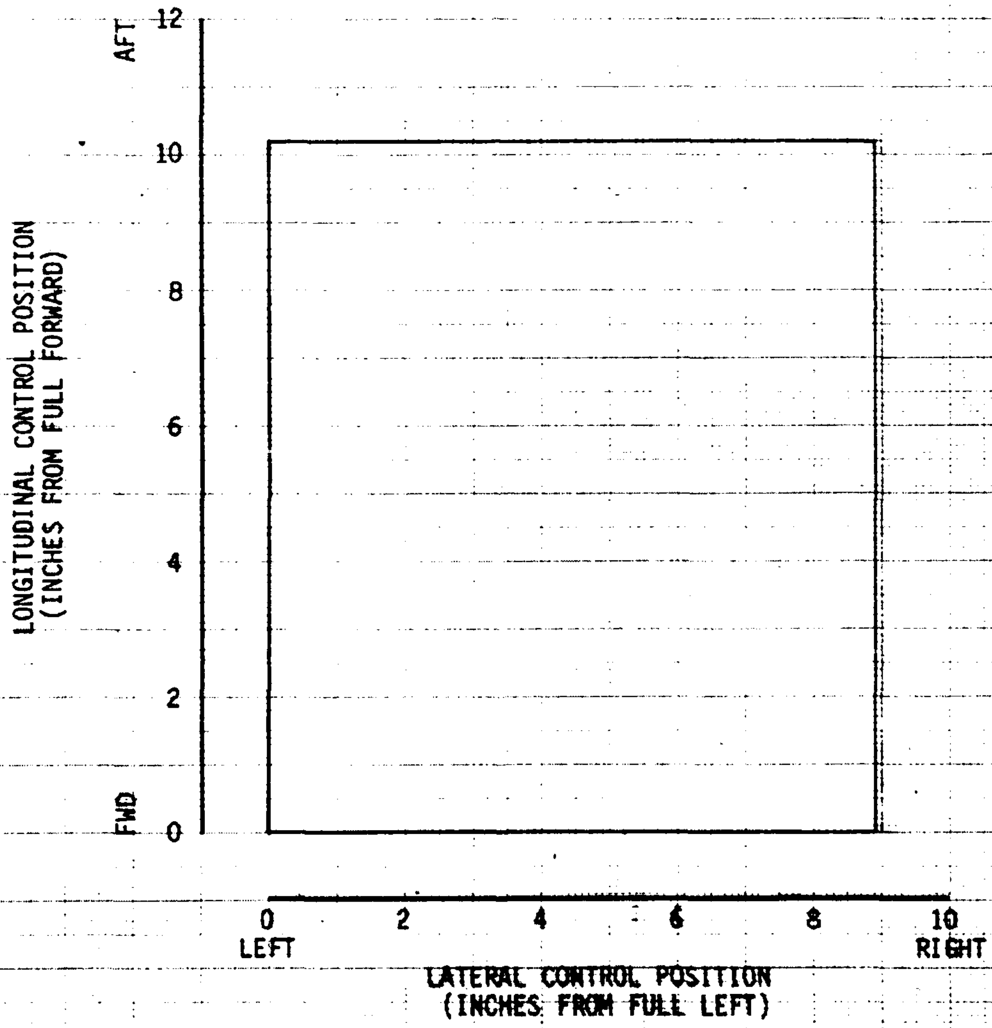


FIGURE 11
LONGITUDINAL CONTROL SYSTEM CHARACTERISTICS
YAH-64 USA S/N 77-23258

NOTES:
1. ROTORS STATIC.
2. FORCES AND POSITIONS MEASURED AT CENTER OF GRIP.
3. HYDRAULIC AND ELECTRICAL POWER PROVIDED BY GROUND POWER UNITS.
4. COPILOT/GUNNER CYCLIC CONTROL RETRACTED.
5. TRIM FEEL ON.

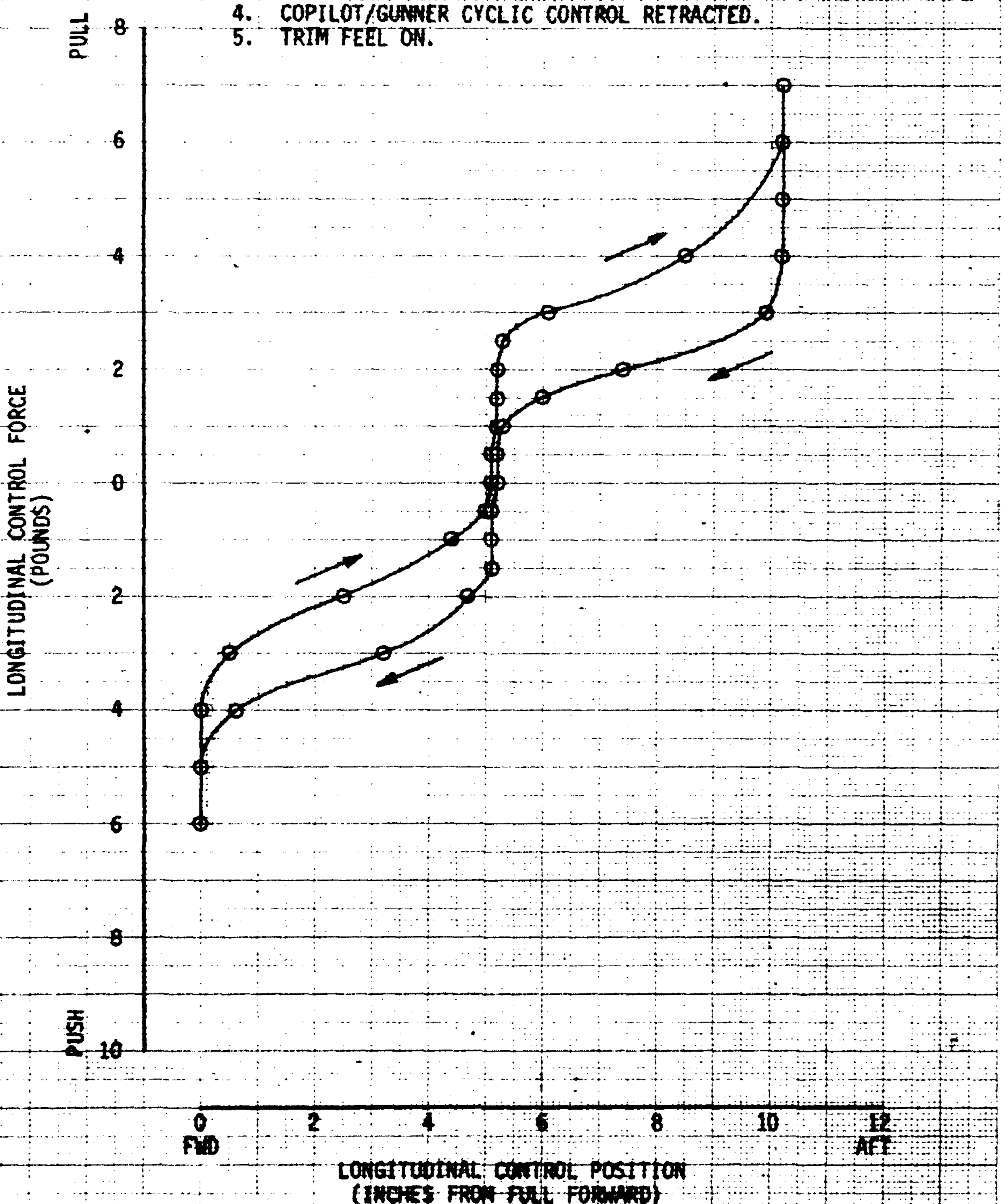


FIGURE 12
LONGITUDINAL CONTROL SYSTEM CHARACTERISTICS
YAH-64 USA S/N 77-23258

NOTES:
1. ROTORS STATIC.
2. FORCES AND POSITIONS MEASURED AT CENTER OF GRIP.
3. HYDRAULIC AND ELECTRICAL POWER PROVIDED BY GROUND POWER UNITS.
4. COPILOT/GUNNER CYCLIC CONTROL EXTENDED.
5. TRIM FEEL ON.

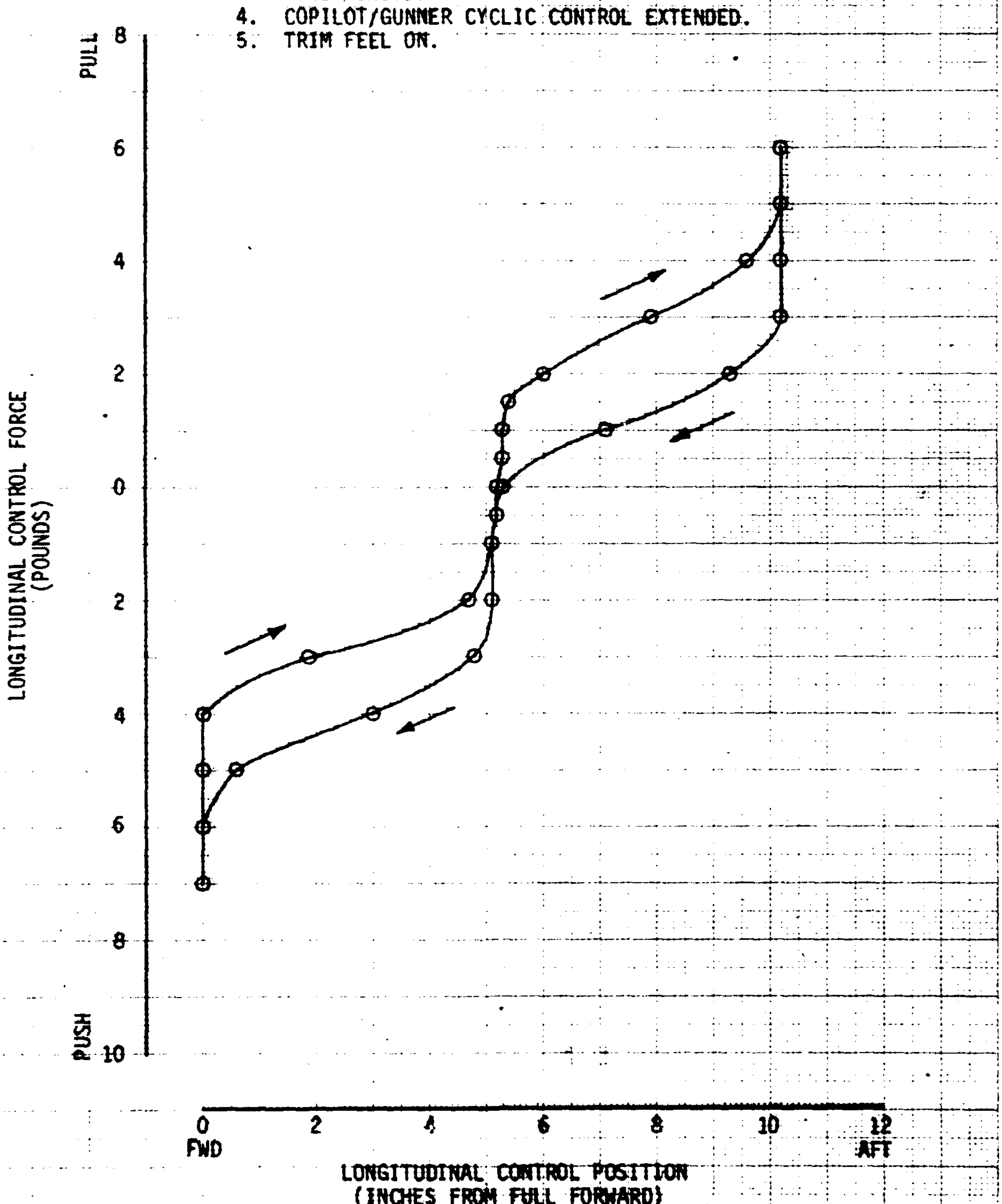


FIGURE 13
LONGITUDINAL CONTROL SYSTEM CHARACTERISTICS
YAH-64 USA S/N 77-23258

NOTES:
1. ROTORS STATIC.
2. FORCES AND POSITIONS MEASURED AT CENTER OF GRIP.
3. HYDRAULIC AND ELECTRICAL POWER PROVIDED BY GROUND POWER UNITS.
4. COPILOT/GUNNER CYCLIC CONTROL RETRACTED.
5. TRIM FEEL OFF.

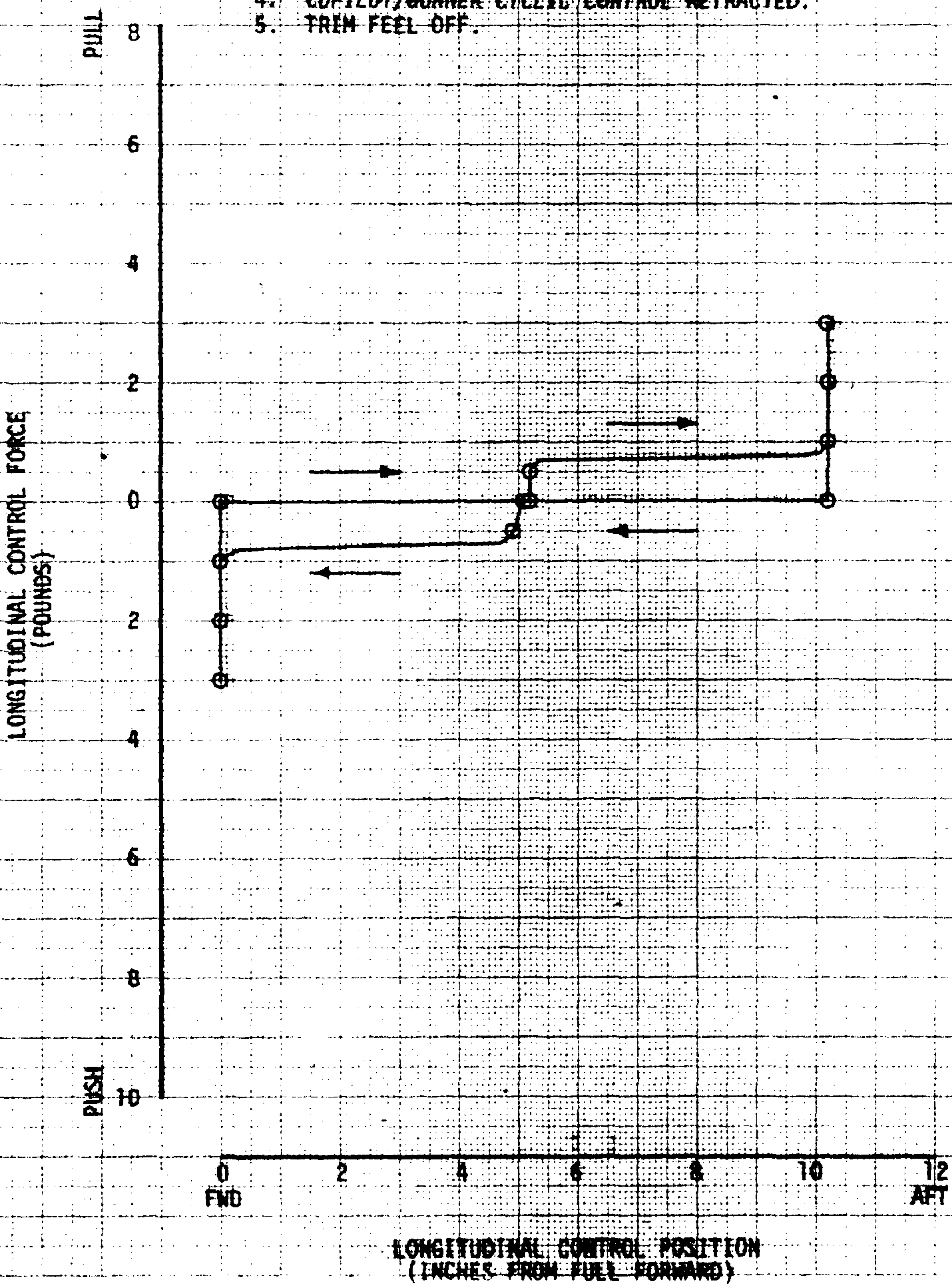


FIGURE 14
LATERAL CONTROL SYSTEM CHARACTERISTICS
YAH-64 USA S/N 77-23258

NOTES:
1. ROTORS STATIC.
2. FORCES AND POSITIONS MEASURED AT CENTER OF GRIP.
3. HYDRAULIC AND ELECTRICAL POWER PROVIDED BY GROUND POWER UNITS.
4. TRIM FEEL ON.

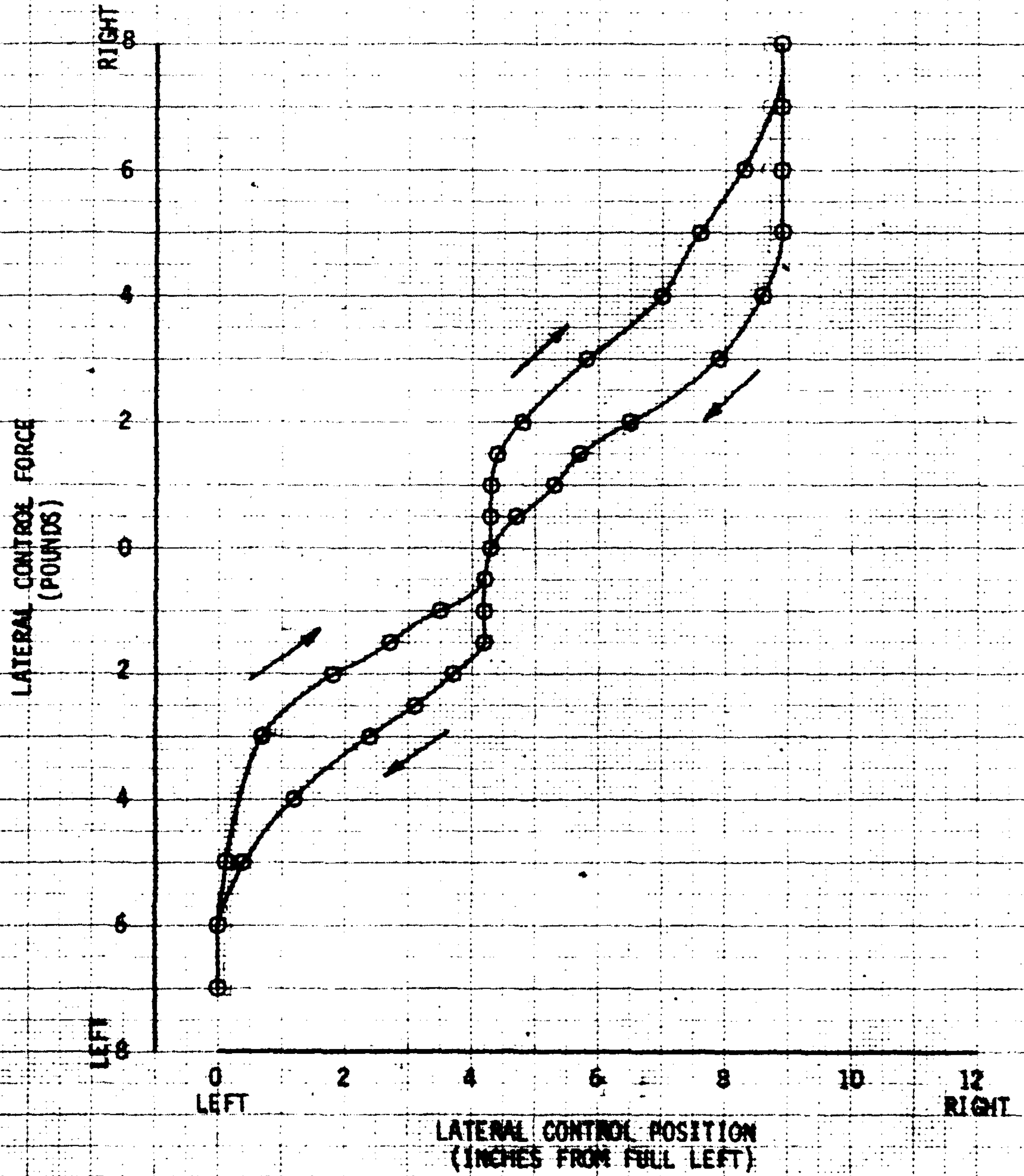


FIGURE 15
LATERAL CONTROL SYSTEM CHARACTERISTICS
YAH-64 USA S/N 77-23258

NOTES: 1. ROTORS STATIC.
2. FORCES AND POSITIONS MEASURED AT CENTER OF GRIP.
3. HYDRAULIC AND ELECTRICAL POWER PROVIDED BY GROUND POWER UNITS.
4. TRIM FEEL OFF.

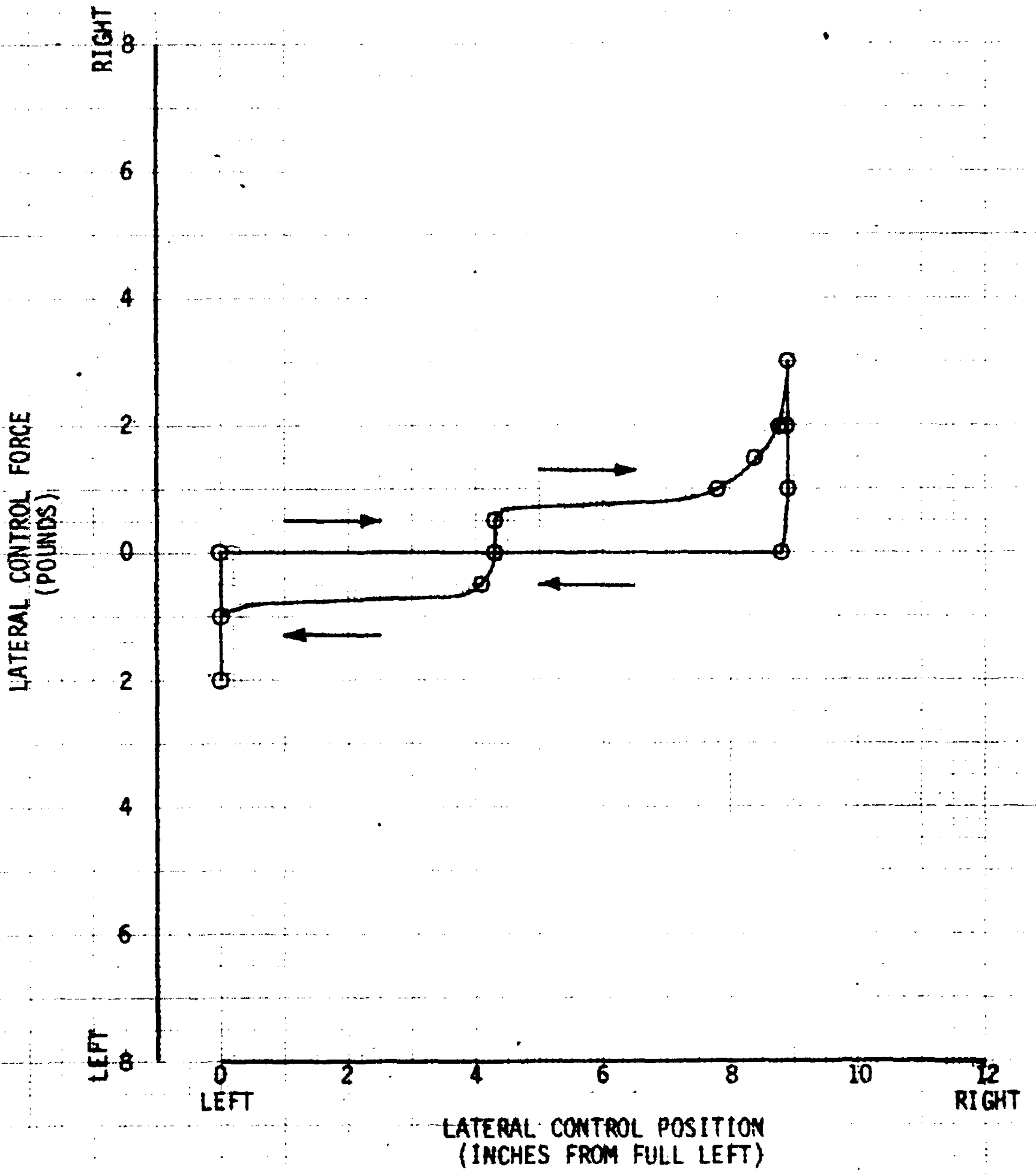


FIGURE 16
DIRECTIONAL CONTROL SYSTEM CHARACTERISITICS
YAH-64 USA S/N 77-23258

NOTES:
1. ROTORS STATIC.
2. FORCES MEASURED AT THE DIRECTIONAL CONTROL.
3. HYDRAULIC AND ELECTRICAL POWER PROVIDED BY GROUND POWER UNITS.
4. TRIM FEEL ON.

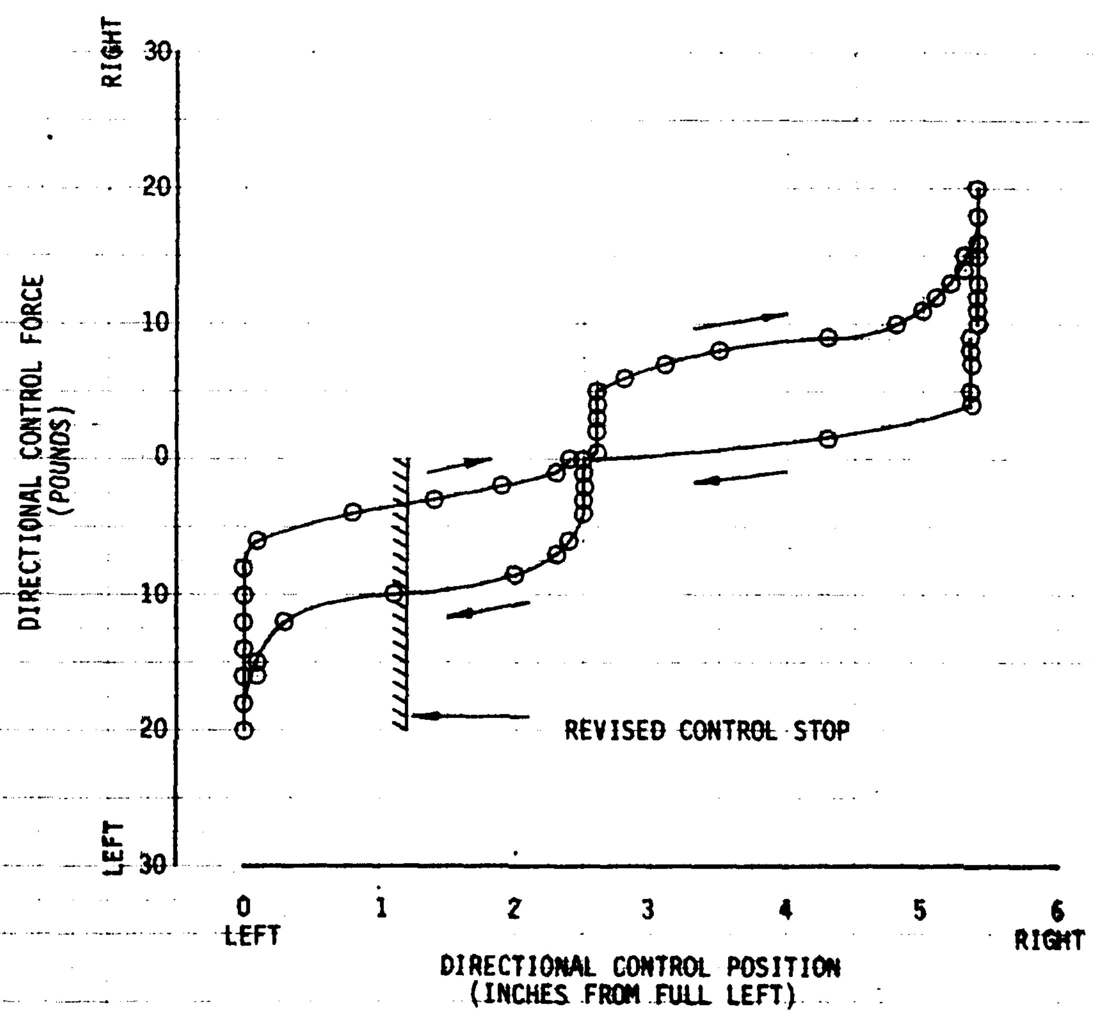
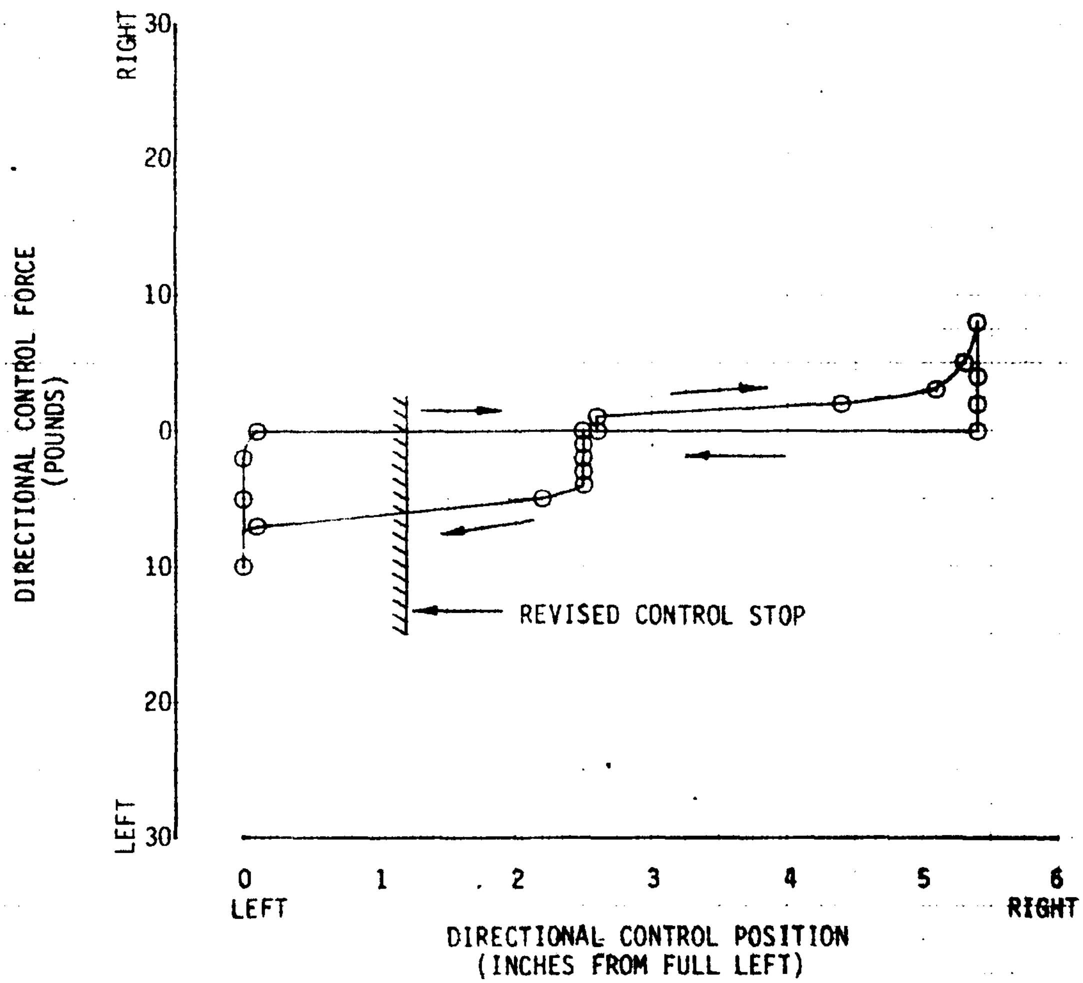


FIGURE 17
DIRECTIONAL CONTROL SYSTEM CHARACTERISTICS
YAH-64 USA S/N 77-23258

NOTES:
1. ROTORS STATIC.
2. FORCES MEASURED AT THE DIRECTIONAL CONTROL.
3. HYDRAULIC AND ELECTRICAL POWER PROVIDED BY GROUND POWER UNITS.
4. TRIM FEEL OFF.
5. REVISED DIRECTIONAL CONTROL STOP INSTALLED AT 1.2 INCHES FROM FULL LEFT.



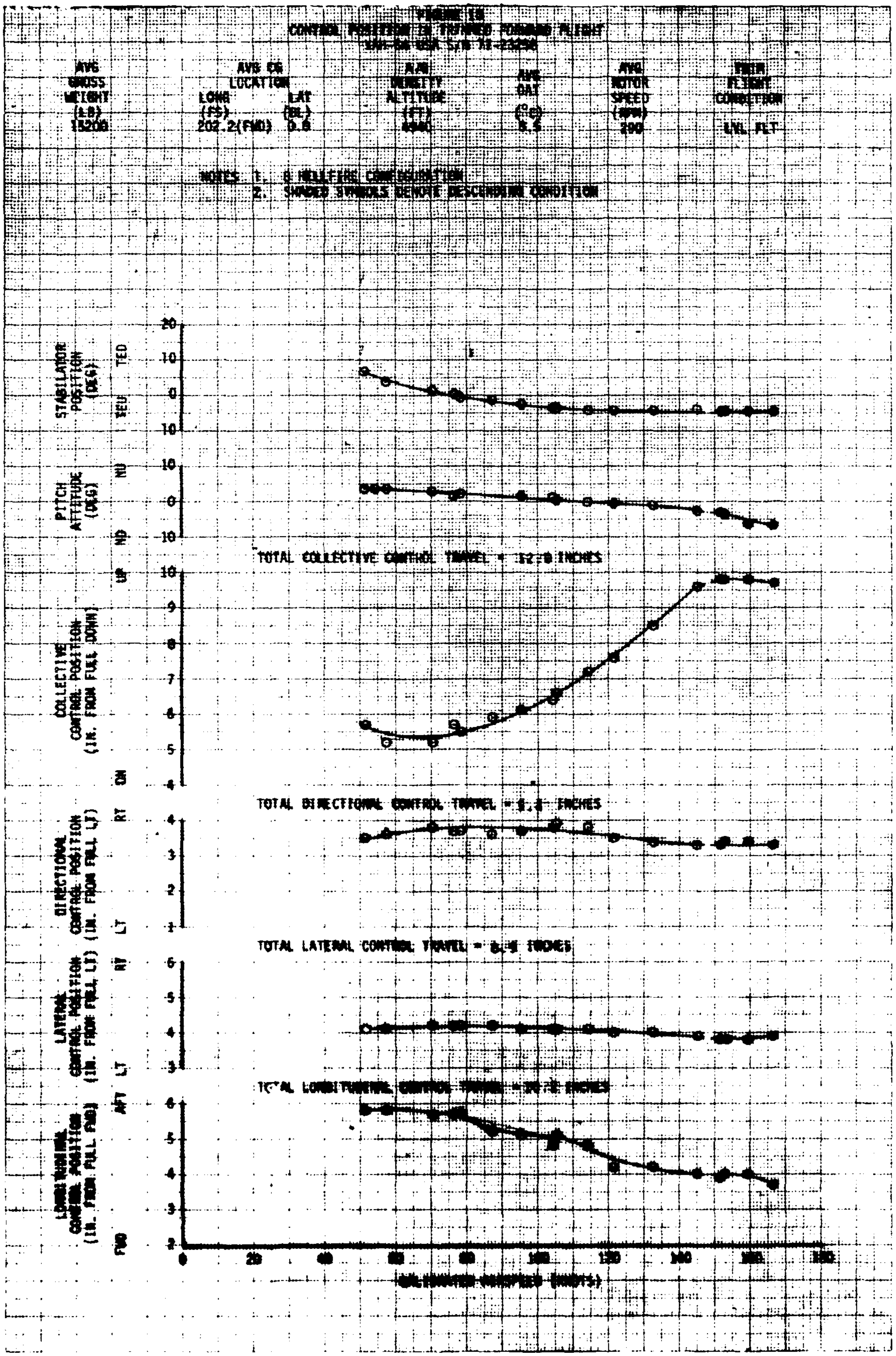

CONTROL POSITIONS IN TRIMMED FORWARD FLIGHT
AVG GROSS WEIGHT (LB) 15200
AVG CG LOCATION LONG (FS) 202.2(FWD) LAT (BL)
DENSITY ALTITUDE (FT)
AVG OAT (°C)
AVG ROTOR SPEED (RPM)
FLIGHT CONDITION LVL FLT
NOTES: 1. HELLFIRE CONFIGURATION
2. SHADED SYMBOLS DENOTE DESCENDING CONDITION
STABILATOR POSITION (DEG) TED TEU
PITCH ATTITUDE (DEG) NU ND
TOTAL COLLECTIVE CONTROL TRAVEL = INCHES
COLLECTIVE CONTROL POSITION (IN. FROM FULL DOWN) UP DN
TOTAL DIRECTIONAL CONTROL TRAVEL = INCHES
DIRECTIONAL CONTROL POSITION (IN. FROM FULL LT) RT LT
TOTAL LATERAL CONTROL TRAVEL = INCHES
LATERAL CONTROL POSITION (IN. FROM FULL LT) RT LT
TOTAL LONGITUDINAL CONTROL TRAVEL = INCHES
LONGITUDINAL CONTROL POSITION (IN. FROM FULL FWD) AFT FWD
CALIBRATED AIRSPEED (KNOTS)

FIGURE 19
CONTROL POSITION IN TRIMMED FORWARD FLIGHT
YAH-64 USA S/N 77-23258

SYMBOL	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION
○	15600	202.3 (FWD)	0.0	5700	8.0	289	CLIMB
□	15580	202.3 (FWD)	0.0	4470	9.0	290	DESCENT

NOTE: 1. 8 HELLFIRE CONFIGURATION.
2. ERRONEOUS AIRSPEED INPUT TO STABILATOR CONTROL UNIT CAUSED STABILATOR FAILURE DURING DESCENT POINTS AT 50 AND 59 KCAS. (SHADED SYMBOLS)
3. CLIMB AT IRP.
4. DESCENTS AT LESS THAN 10% TORQUE PER ENGINE.

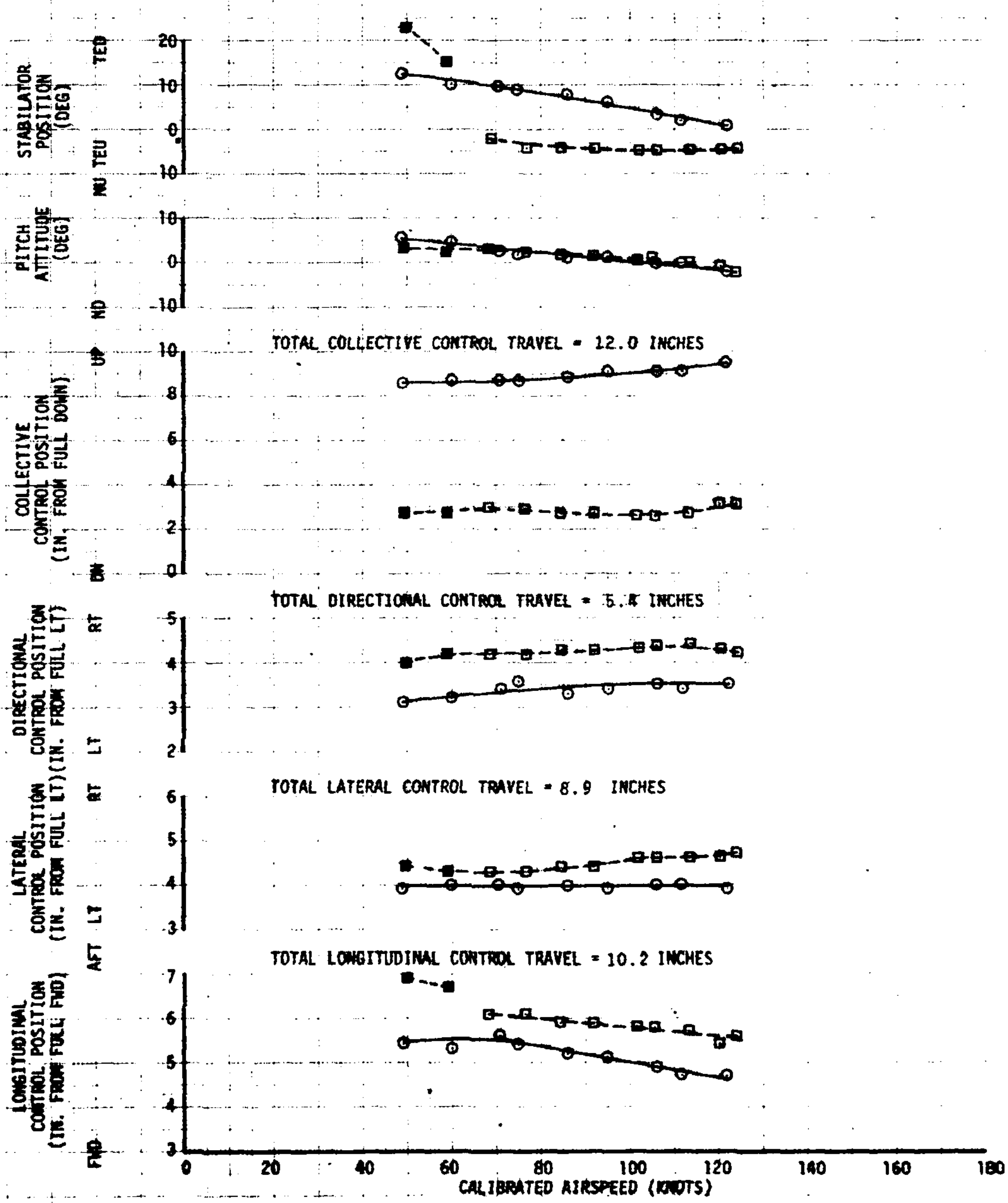


FIGURE 20
COLLECTIVE-FIXED STATIC LONGITUDINAL STABILITY
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	DASE CONDITION
14260	205.8(AFT)	0.0	5200	12	289	LEVEL FLIGHT	ON

NOTES: 1. 8 HELLFIRE CONFIGURATION
2. SHADED SYMBOLS DENOTE TRIM

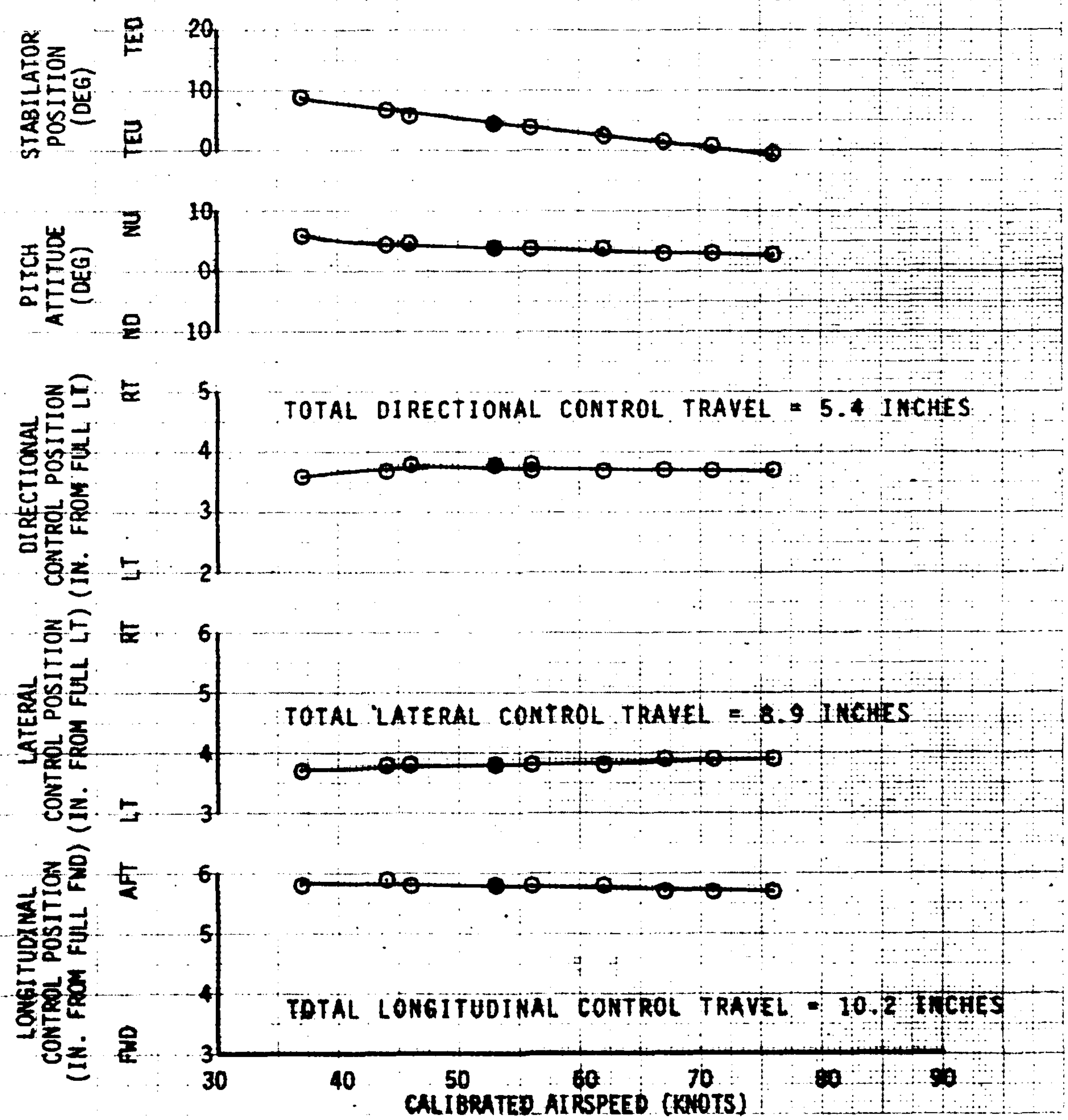


FIGURE 21
COLLECTIVE-FIXED STATIC LONGITUDINAL STABILITY
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	DASE CONDITION
14500	205.9(AFT)	0.0	5200	7.5	289	LVL FLT	ON

NOTES: 1. 8-HELLFIRE CONFIGURATION
2. SHADED SYMBOLS DENOTE TRIM

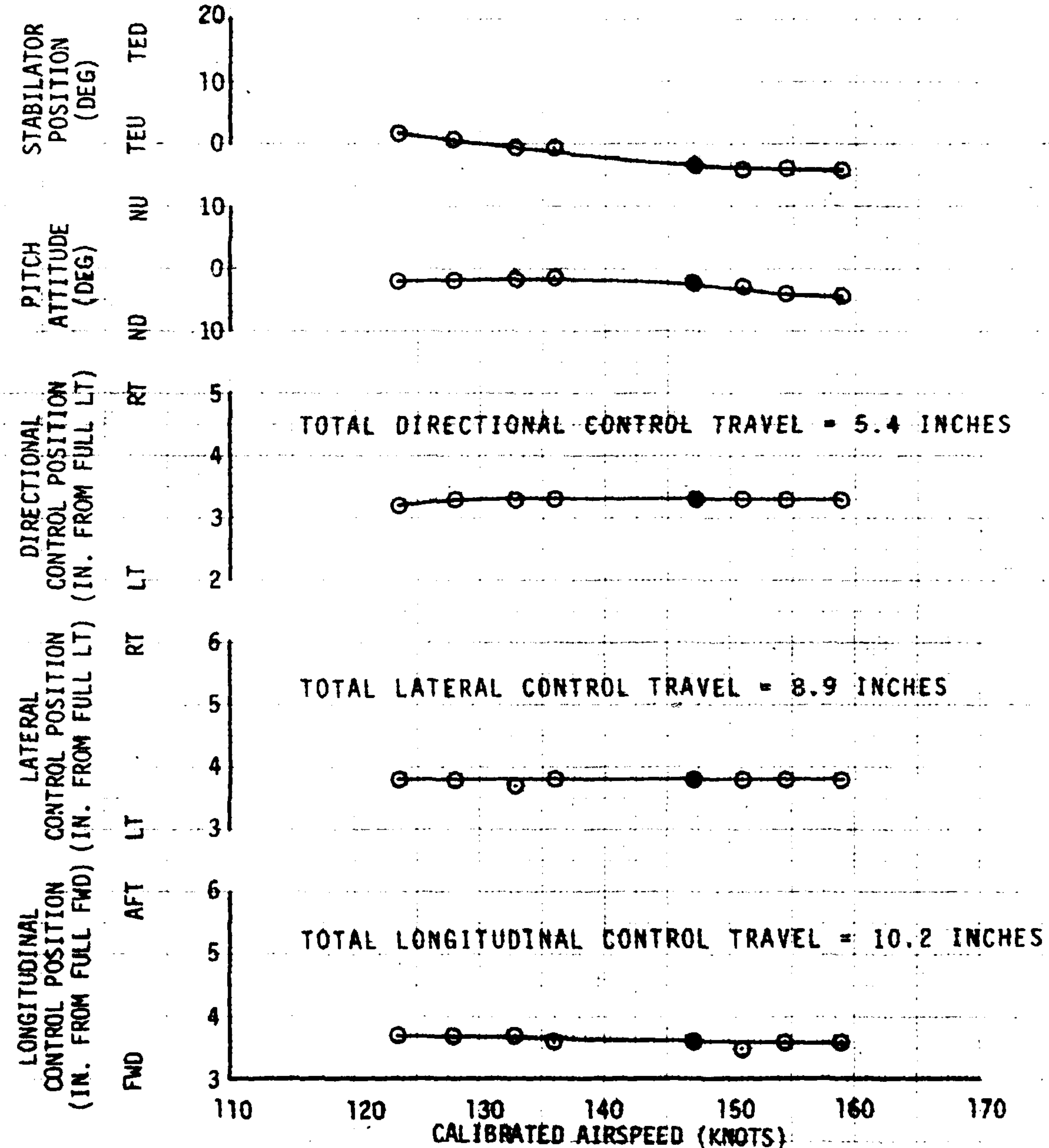


FIGURE 22
STATIC LATERAL-DIRECTIONAL STABILITY
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KCAS)	DASE CONDITION
14500	205.7(AFT)	0.0	4400	11.5	289	56	ON

NOTES: 1. 8-HELLFIRE CONFIGURATION
2. SHADED SYMBOLS DENOTE TRIM

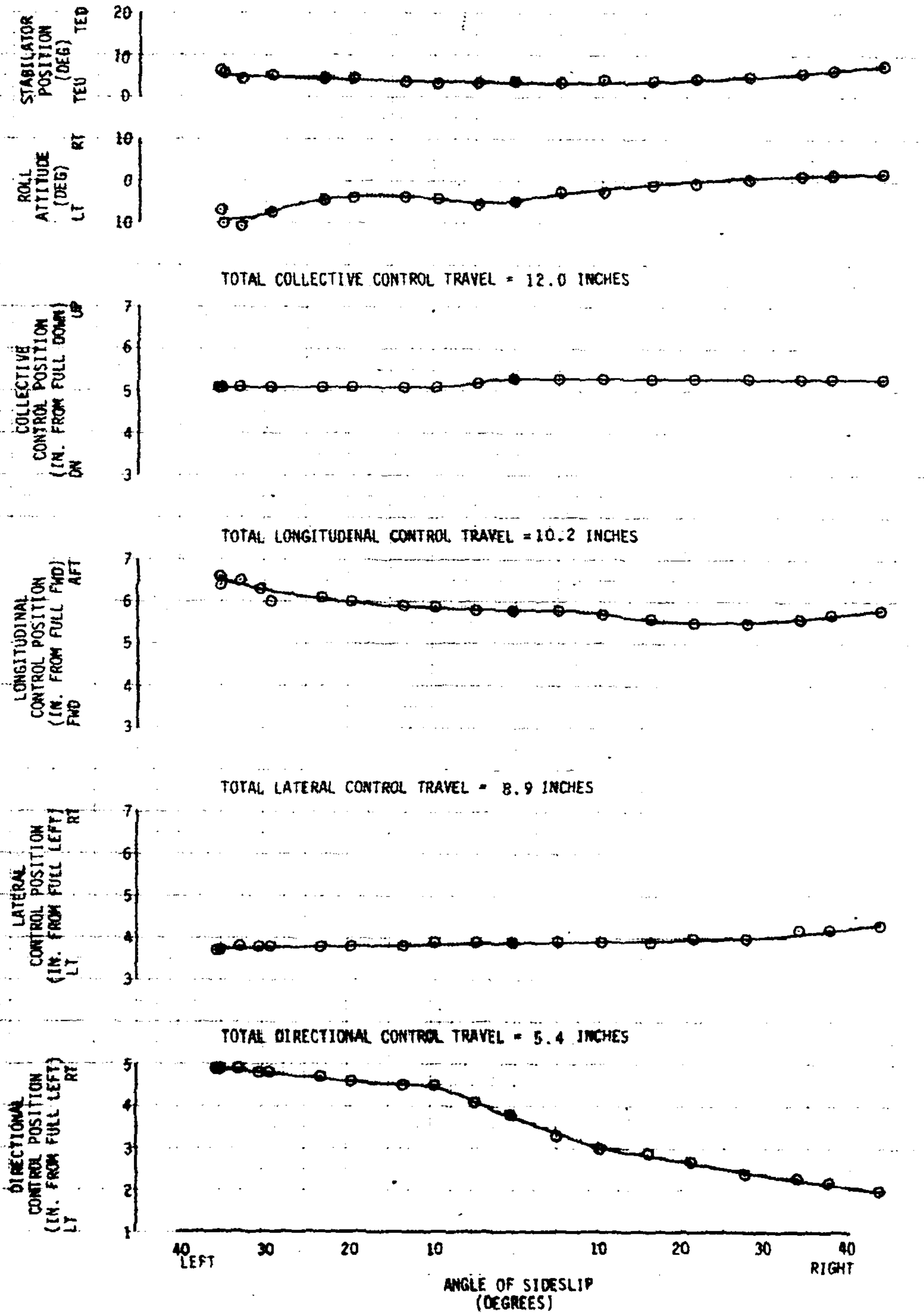


FIGURE 23
STATIC LATERAL - DIRECTIONAL STABILITY
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG C.G. LOCATION LONG (FS)	AVG C.G. LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KCAS)	DASE CONDITION
14740	205.8(AFT)	0.0	5440	6.0	289	144.0	ON

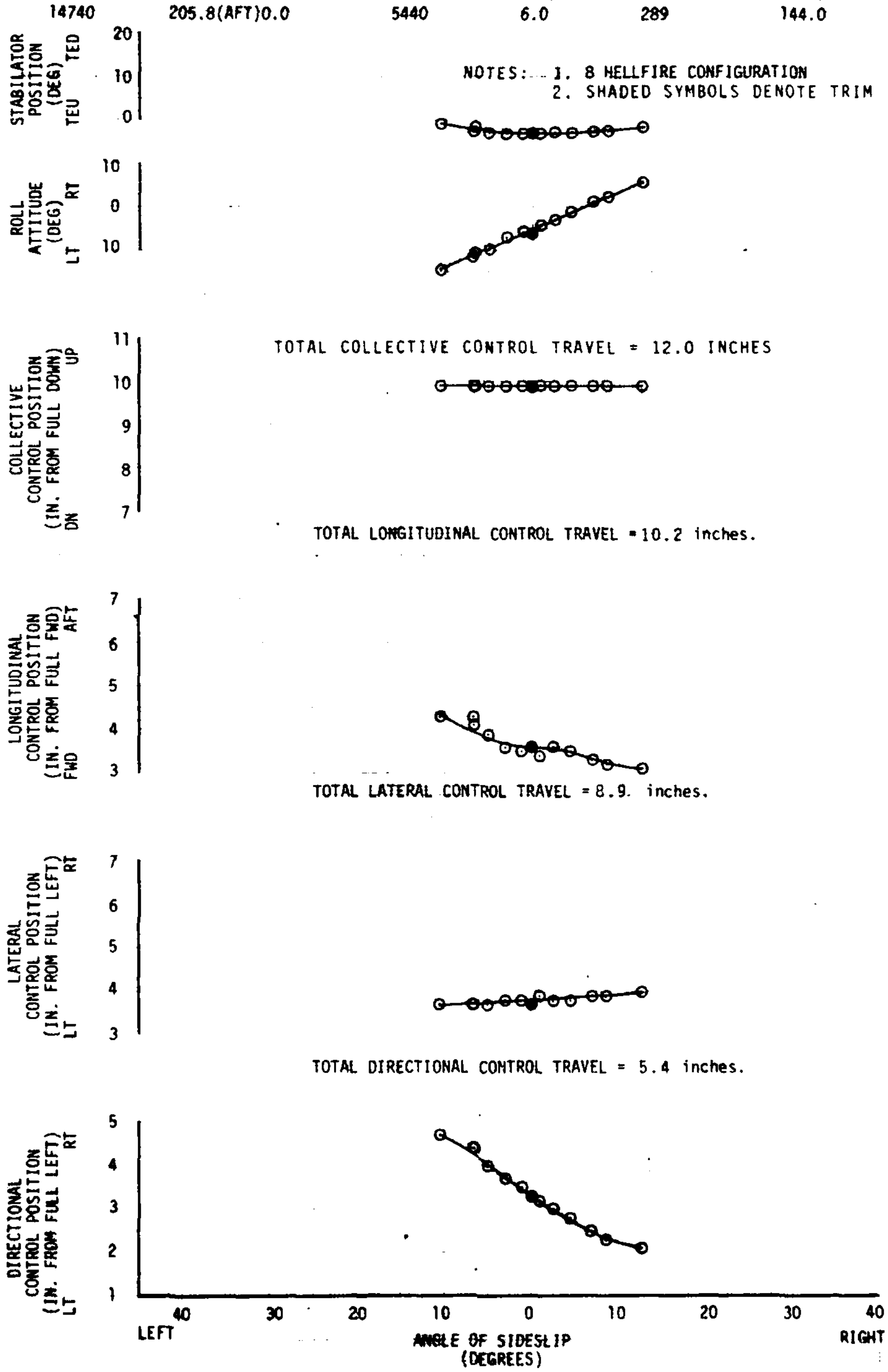


FIGURE 24
MANEUVERING STABILITY
YAH-64 USA S/N 77-23258

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	DASE CONDITION
—○—	14720	206.0	0.0	5320	6.5	289	RT TURN	ON
--□--	14920	205.8	0.0	5580	6.5	289	LT TURN	ON

NOTES: 1. 8 HELLFIRE CONFIGURATION.
2. 145 KCAS.

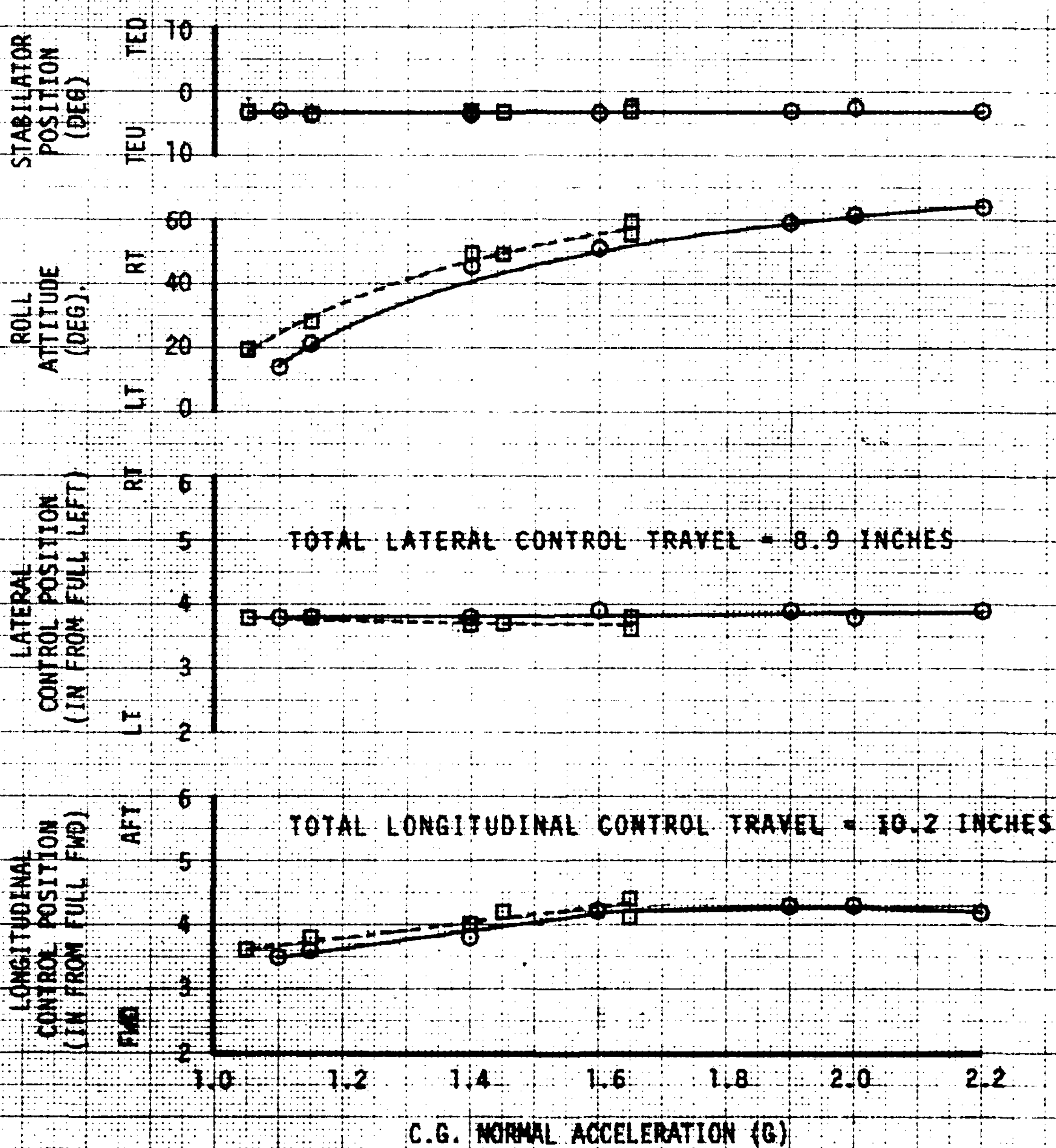


FIGURE 25
MANEUVERING STABILITY
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (C)	AVG ROTOR SPEED (RPM)	AVG CALIBRATED AIRSPEED (KT)	DASE CONDITION	CONFIG.
14660	206.1(AFT)	0.0	5620	6.0	290	146	ON	3-HELLFIRE

NOTE: DATA OBTAINED DURING STEADY TURN

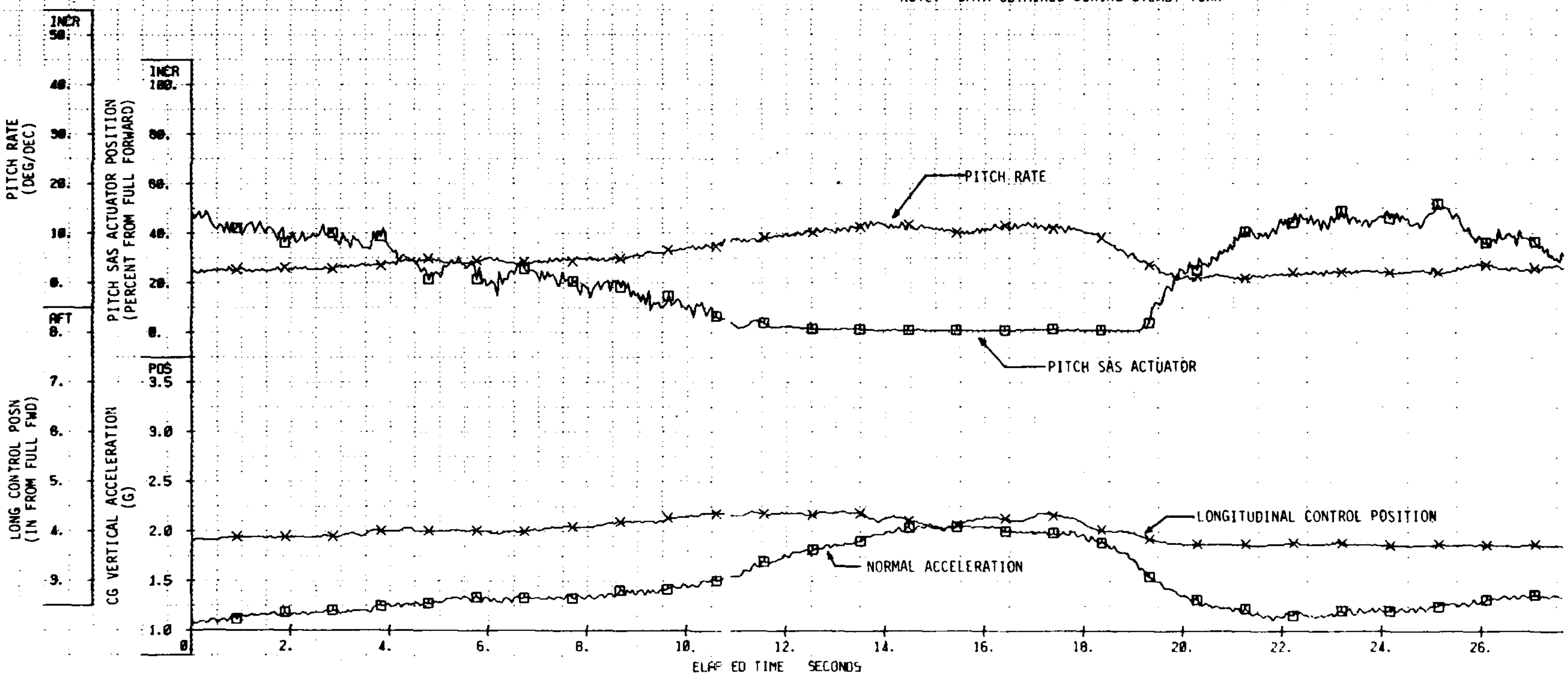


FIGURE 26
3-AXIS SHORT-TERM OSCILLATION
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	DASE CONDITION	CONFIG
15040	206.5(AFT)	0.0	5620	6.5	290	144	OFF	8-HELLFIRE

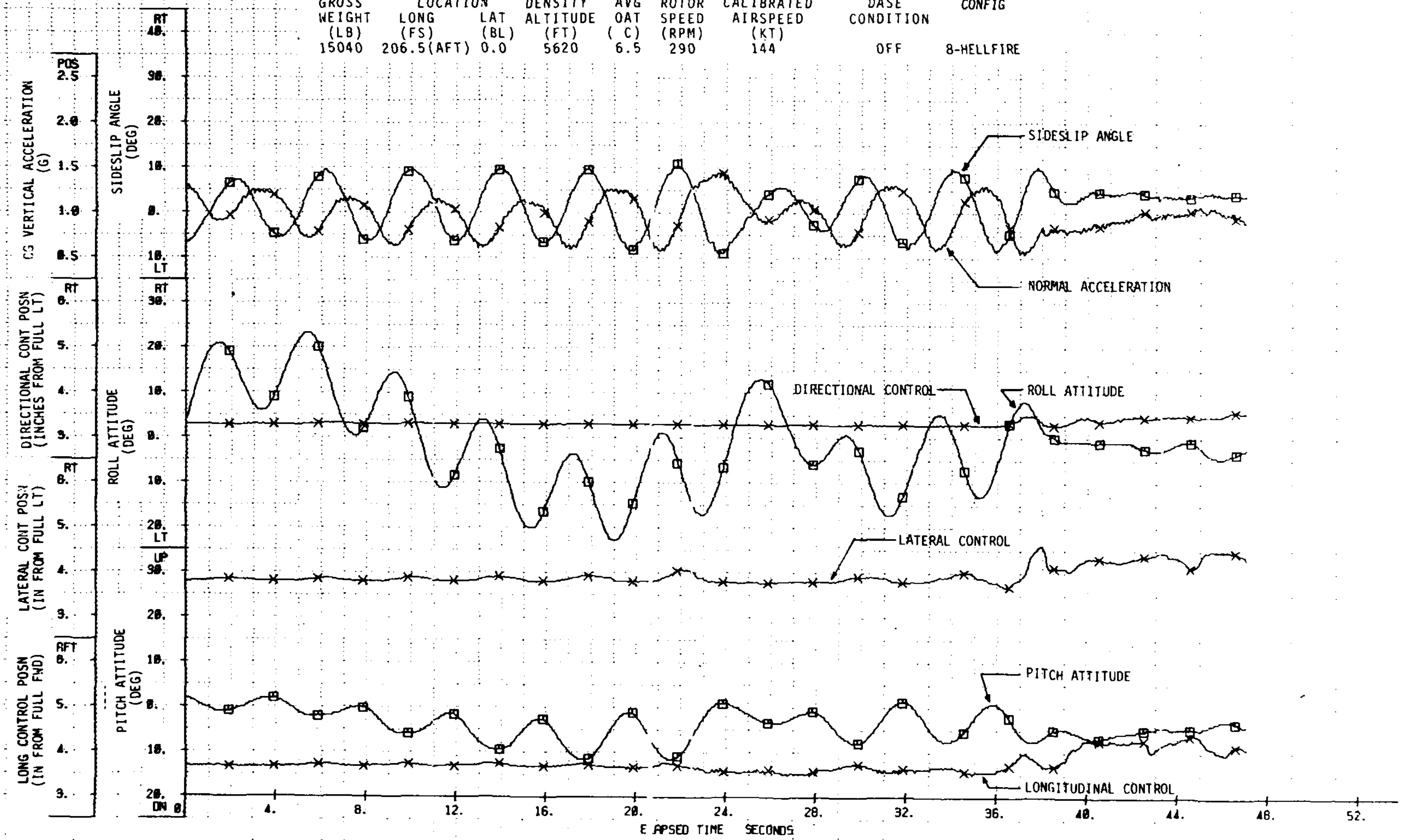


FIGURE 27
LONGITUDINAL LONG-PERIOD RESPONSE
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (C)	AVG ROTOR SPEED (RPM)	DASE CONDITION	CONFIGURATION
14340	205.9(AFT)	0.0	4660	8.5	290	ON	8-HELLFIRE

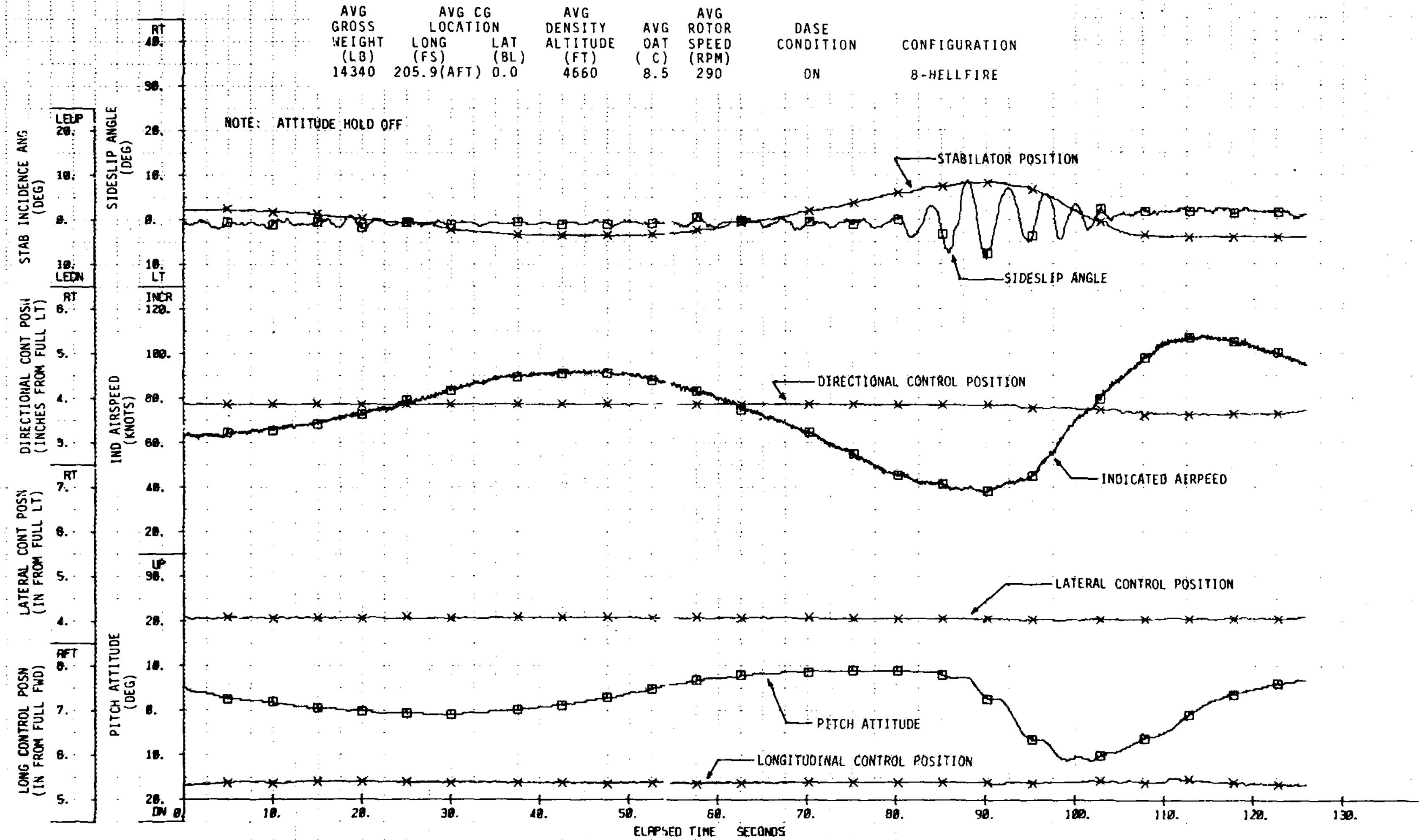
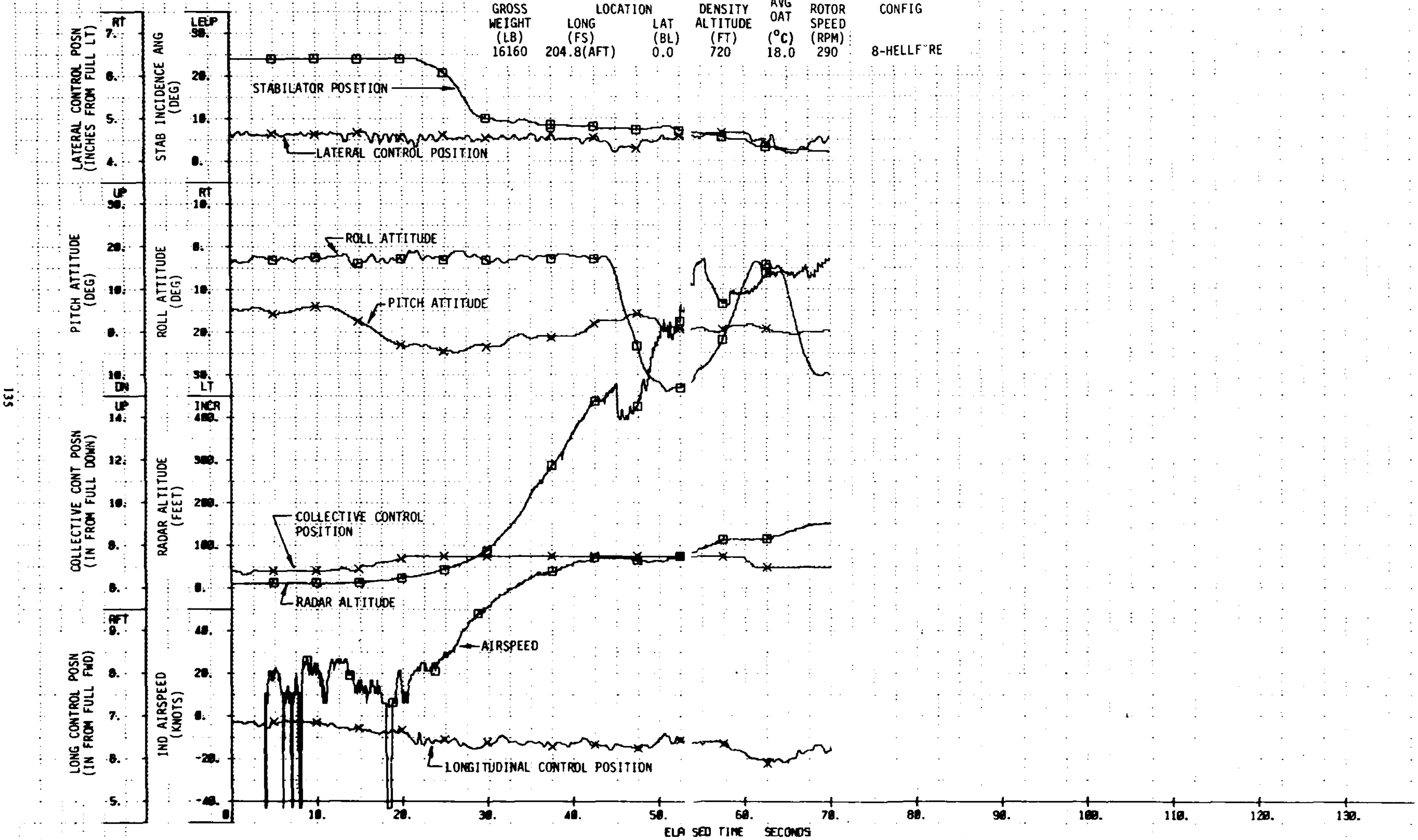


FIGURE 28
NORMAL TAKEOFF
YAH-64 USA S/N 77-23258

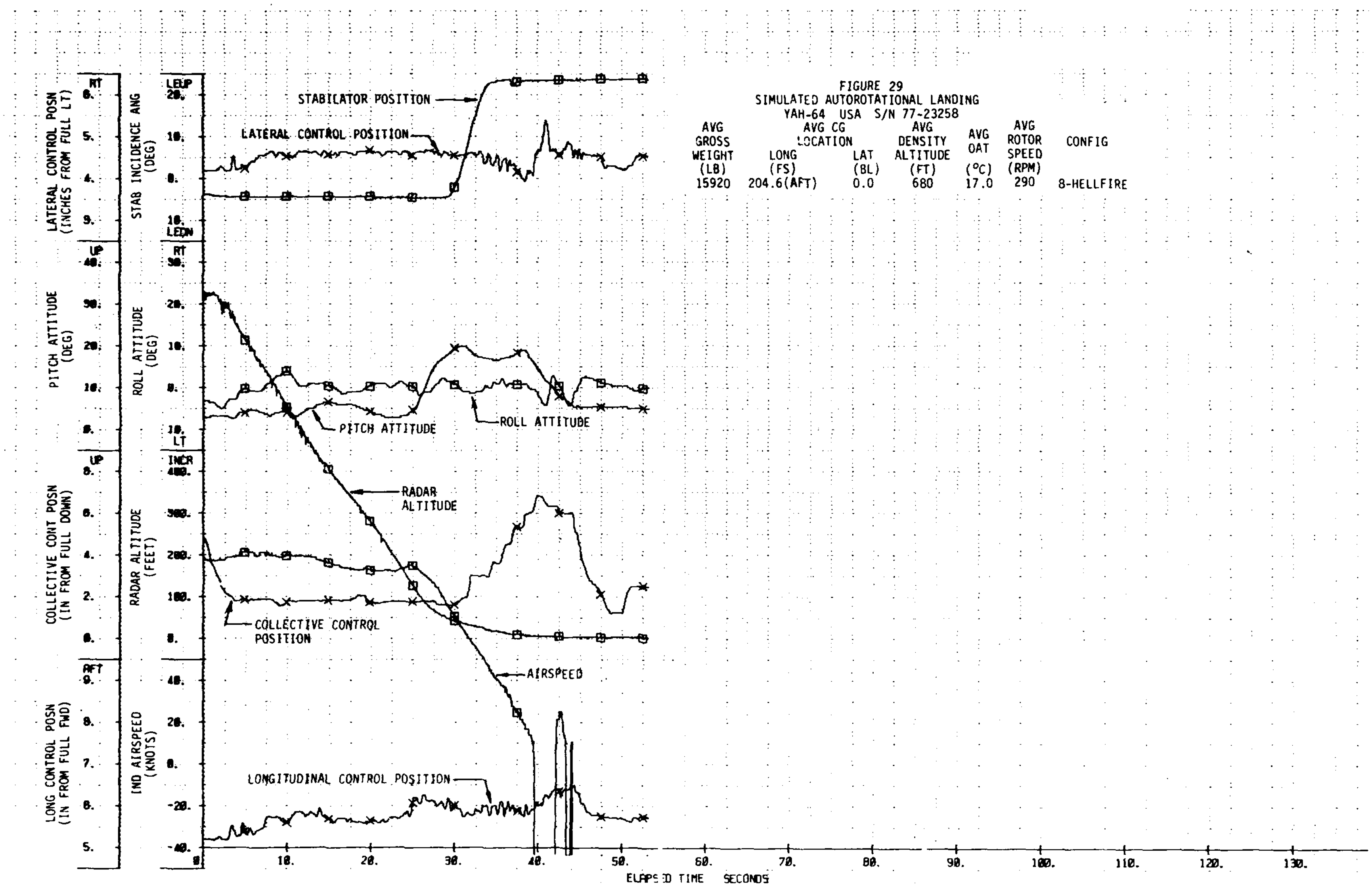
AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	CONFIG
16160	204.8(AFT)	0.0	720	18.0	290	8-HELLF"RE



445

FIGURE 29
SIMULATED AUTOROTATIONAL LANDING
YAH-64 USA S/N 77-23258

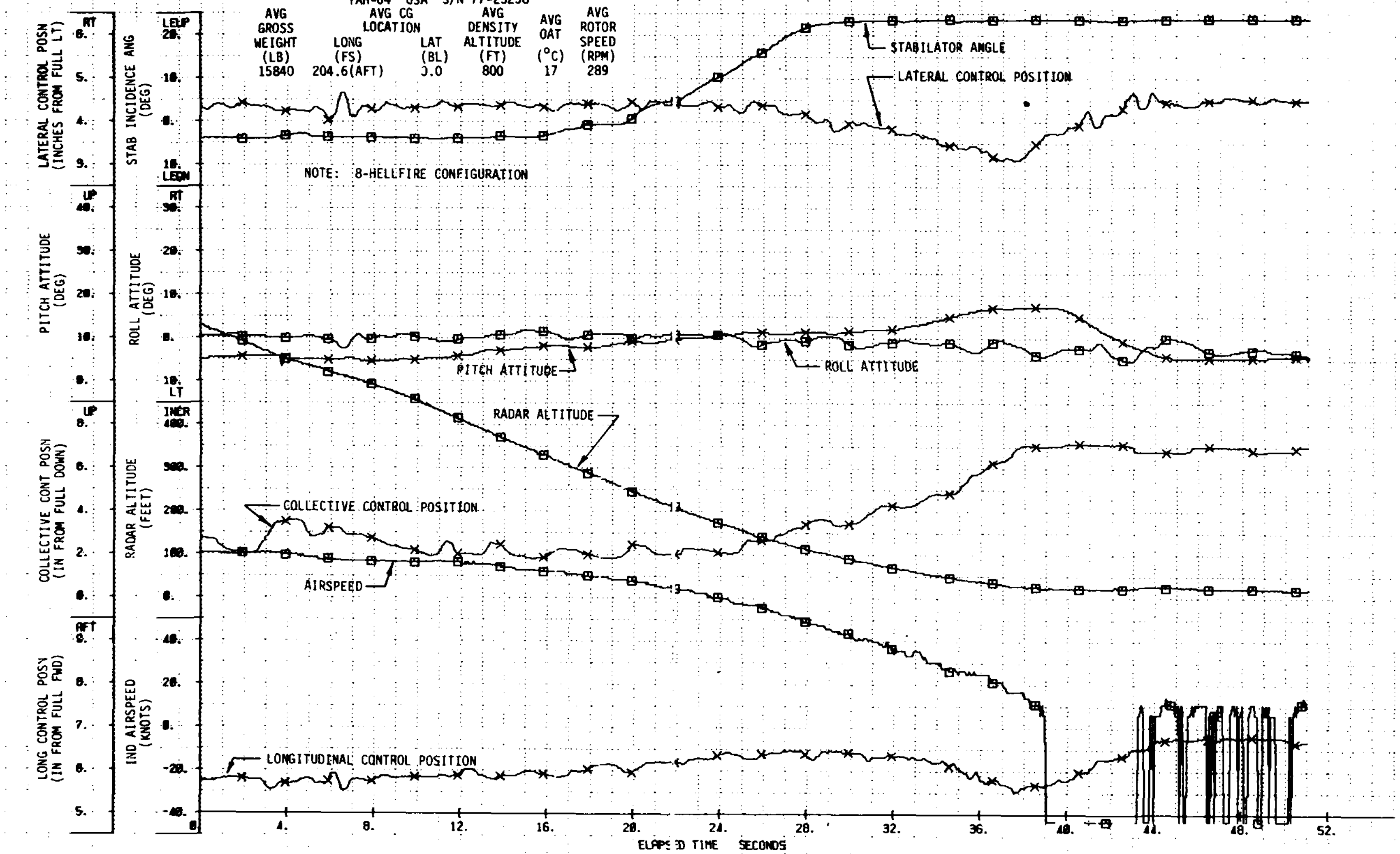
AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	CONFIG
15920	204.6(AFT)	0.0	680	17.0	290	8-HELLFIRE



432

431

FIGURE 30
STEEP APPROACH
YAH-64 USA S/N 77-23258



424

423

FIGURE 31
LOW-SPEED FORWARD AND REARWARD FLIGHT
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	WHEEL HEIGHT (FT)	DASE CONDITION
14720	201.4(FWD)	0.0	280	14.0	289	20	ON

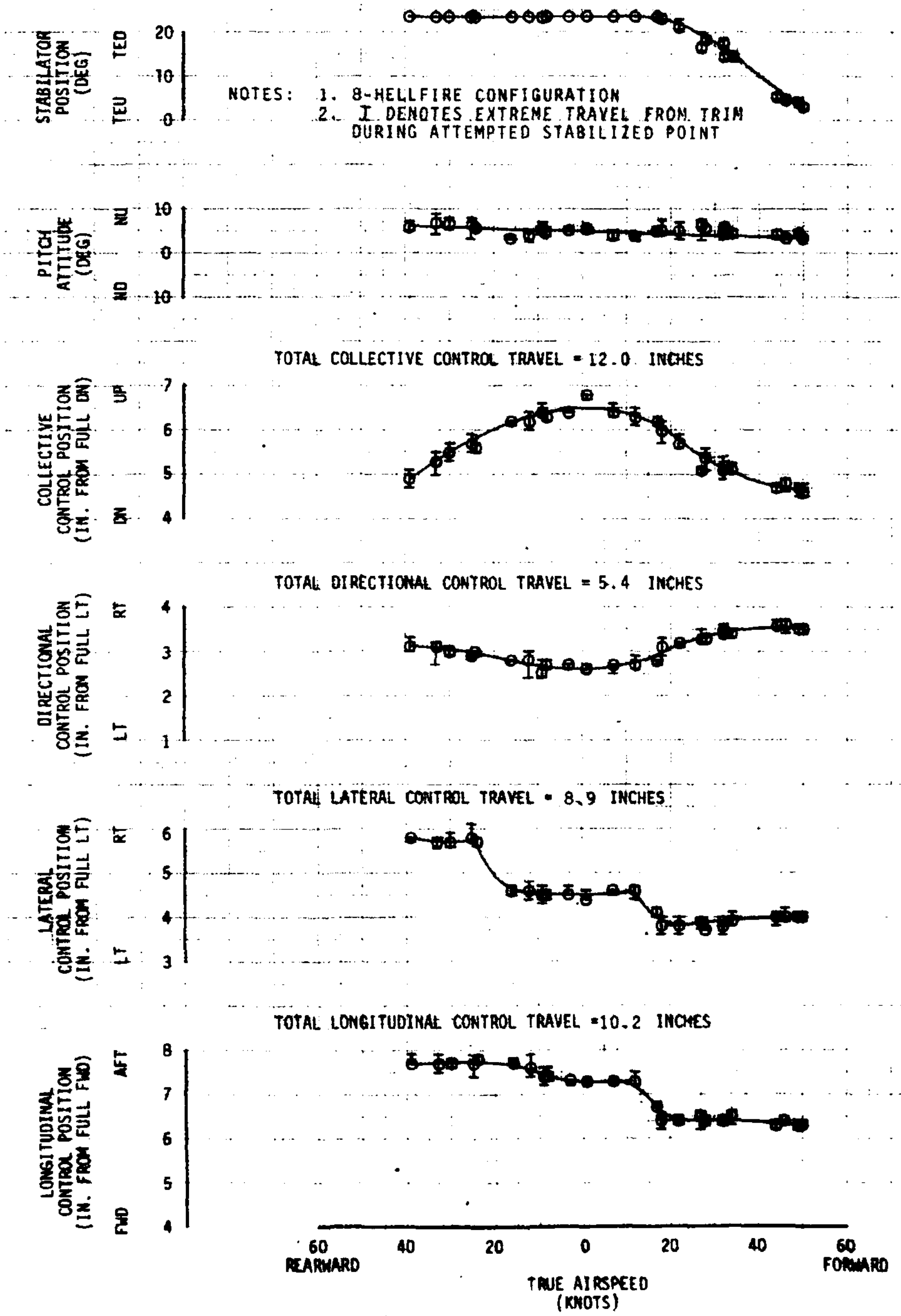
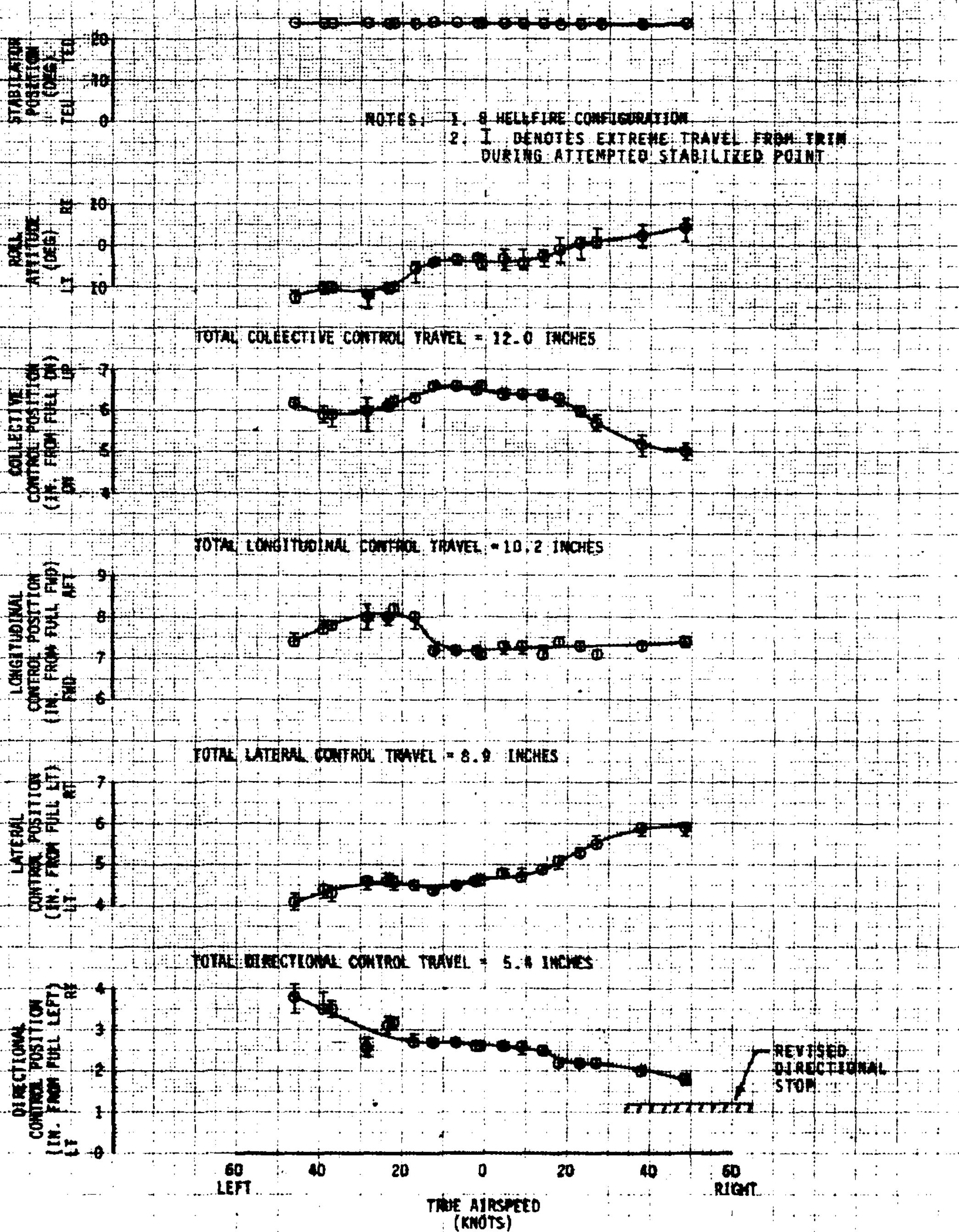
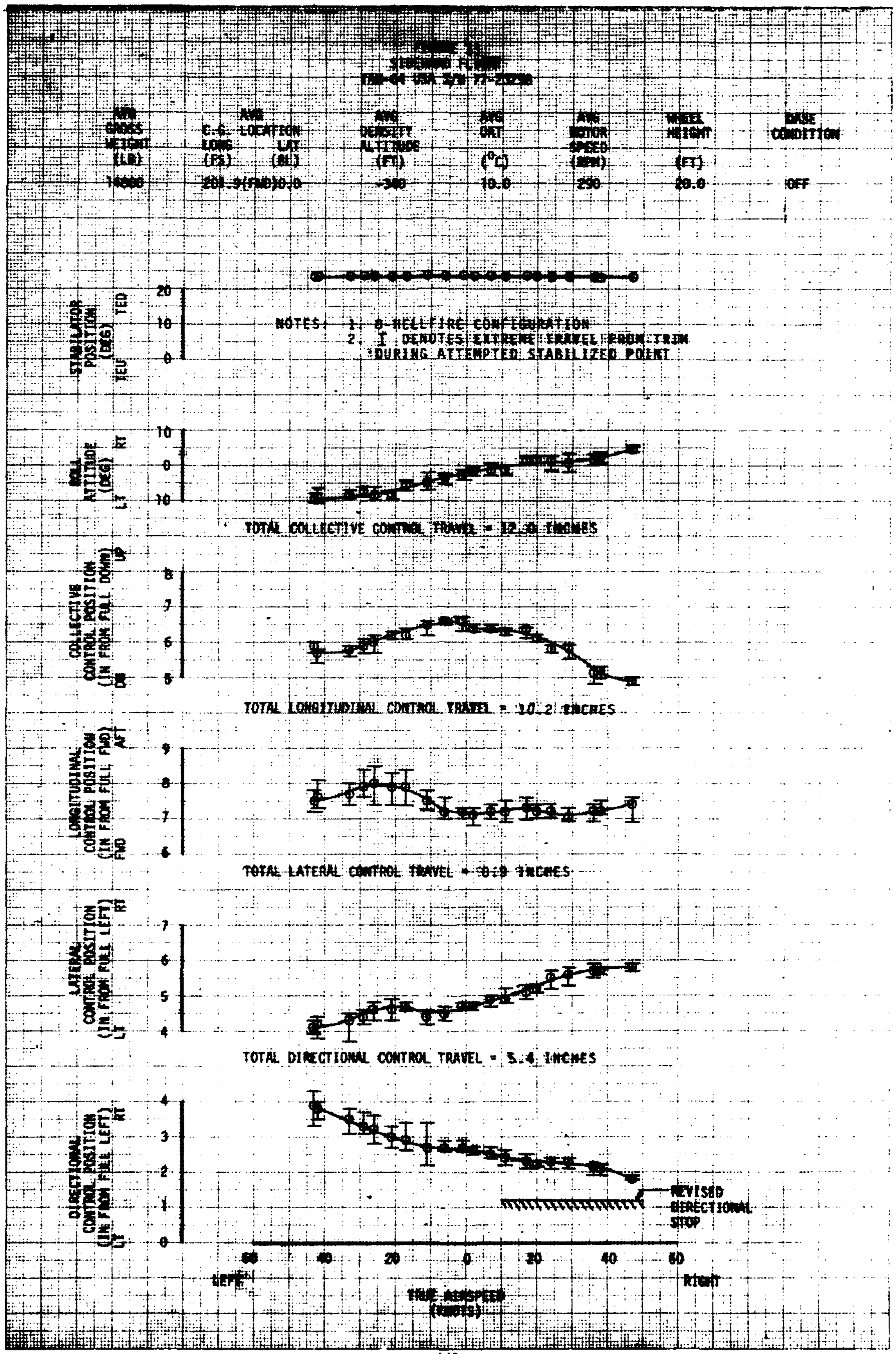


FIGURE 32
SIDEWARD FLIGHT
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	WHEEL HEIGHT (FT)	BASE CONDITION
14820	203.3(FWD)	0.0	440	15.0	289	20	OGE





AVG GROSS WEIGHT (LB)
AVG C.G. LOCATION LONG (FS) LAT (BL)
AVG DENSITY ALTITUDE (FT)
AVG OAT (°C)
AVG ROTOR SPEED (RPM)
WHEEL HEIGHT (FT)
BASE CONDITION
14000
201.9(FWD)0.0
-300
10.0
290
20.0
OFF
NOTES: 1. 8-HELLFIRE CONFIGURATION
2. I DENOTES EXTREME TRAVEL FROM TRIM DURING ATTEMPTED STABILIZED POINT
STABILATOR POSITION (DEG)
TED
TEU
ROLL ATTITUDE (DEG)
RT
LT
TOTAL COLLECTIVE CONTROL TRAVEL = 12.30 INCHES
COLLECTIVE CONTROL POSITION (IN FROM FULL DOWN)
UP
DN
TOTAL LONGITUDINAL CONTROL TRAVEL = 10.2 INCHES
LONGITUDINAL CONTROL POSITION (IN FROM FULL FWD)
AFT
FWD
TOTAL LATERAL CONTROL TRAVEL = 8.9 INCHES
LATERAL CONTROL POSITION (IN FROM FULL LEFT)
RT
LT
TOTAL DIRECTIONAL CONTROL TRAVEL = 5.4 INCHES
DIRECTIONAL CONTROL POSITION (IN FROM FULL LEFT)
RT
LT
REVISED DIRECTIONAL STOP
LEFT
RIGHT
TRUE AIRSPEED (KNOTS)

FIGURE 34
CRITICAL AZIMUTH
YAH-64 USA S/N 77-23258

	AVG GROSS WEIGHT (LB)	AVG C.G. LOCATION LONG (FS)	LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	WHEEL HEIGHT (FT)	TRIM TRUE AIRSPEED (KTS)	DASE CONDITION
○	14420	201.3(FWD)	0.0	400	15.0	20	30.0	ON
□	14840	201.8(FWD)	0.0	-140	11.5	20	30.0	ON

NOTES 1. 8 HELLFIRE CONFIGURATION
2. I DENOTES EXTREME TRAVEL FROM TRIM DURING ATTEMPTED STABILIZED POINT

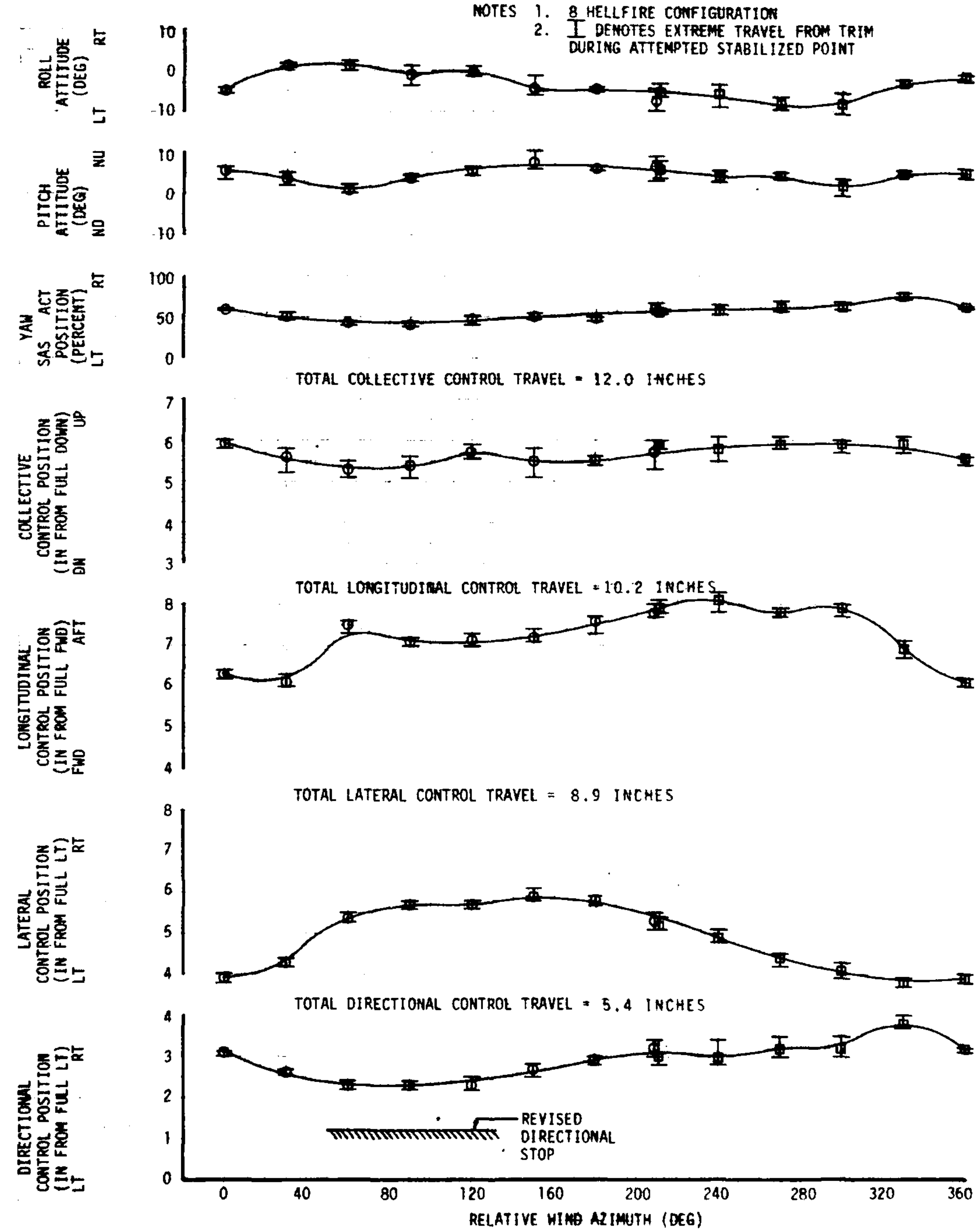


FIGURE 35
CRITICAL AZIMUTH
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG C.G. LOCATION LONG (FS)	AVG C.G. LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	WHEEL HEIGHT (FT)	BASE CONDITION
14530	201.7(FWD)	0.0	10	12.0	290	20	ON

NOTES:
1. 8 HELLFIRE CONFIGURATION
2. RELATIVE WIND AZIMUTH 240 DEG.
3. Ɪ DENOTES EXTREME TRAVEL FROM TRIM DURING ATTEMPTED STABILIZED POINT

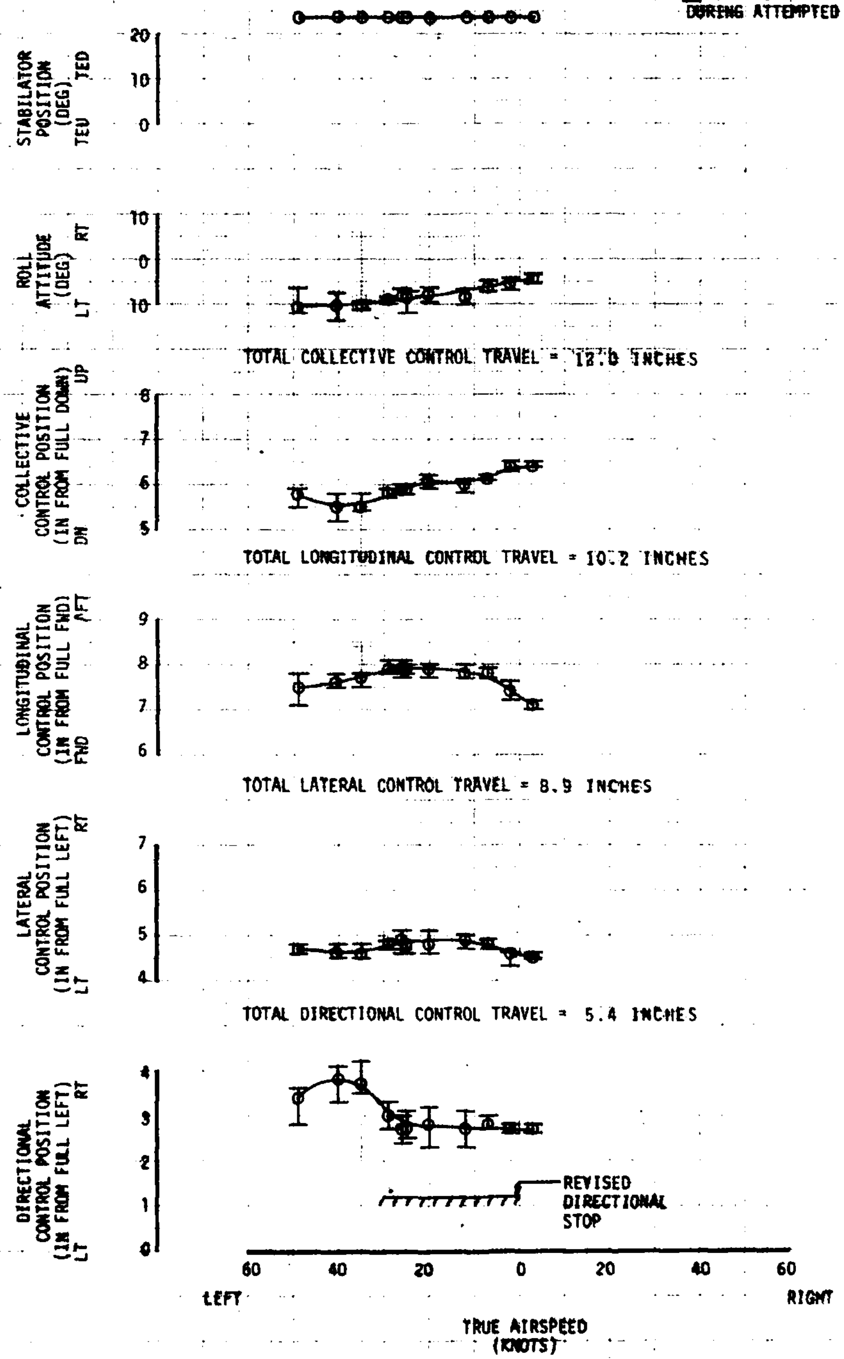


FIGURE 36
CRITICAL AZIMUTH
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG C.G. LOCATION LONG (FS)	AVG C.G. LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	WHEEL HEIGHT (FT)	DASE CONDITION
14500	201.9(FWD)	0.0	-180	11.5	290	20.0	OFF

NOTES: 1. 8 HELLFIRE CONFIGURATION
2. RELATIVE WIND AZIMUTH 240 DEG.
3. I DENOTES EXTREME TRAVEL FROM TRIM DURING ATTEMPTED STABILIZED POINT

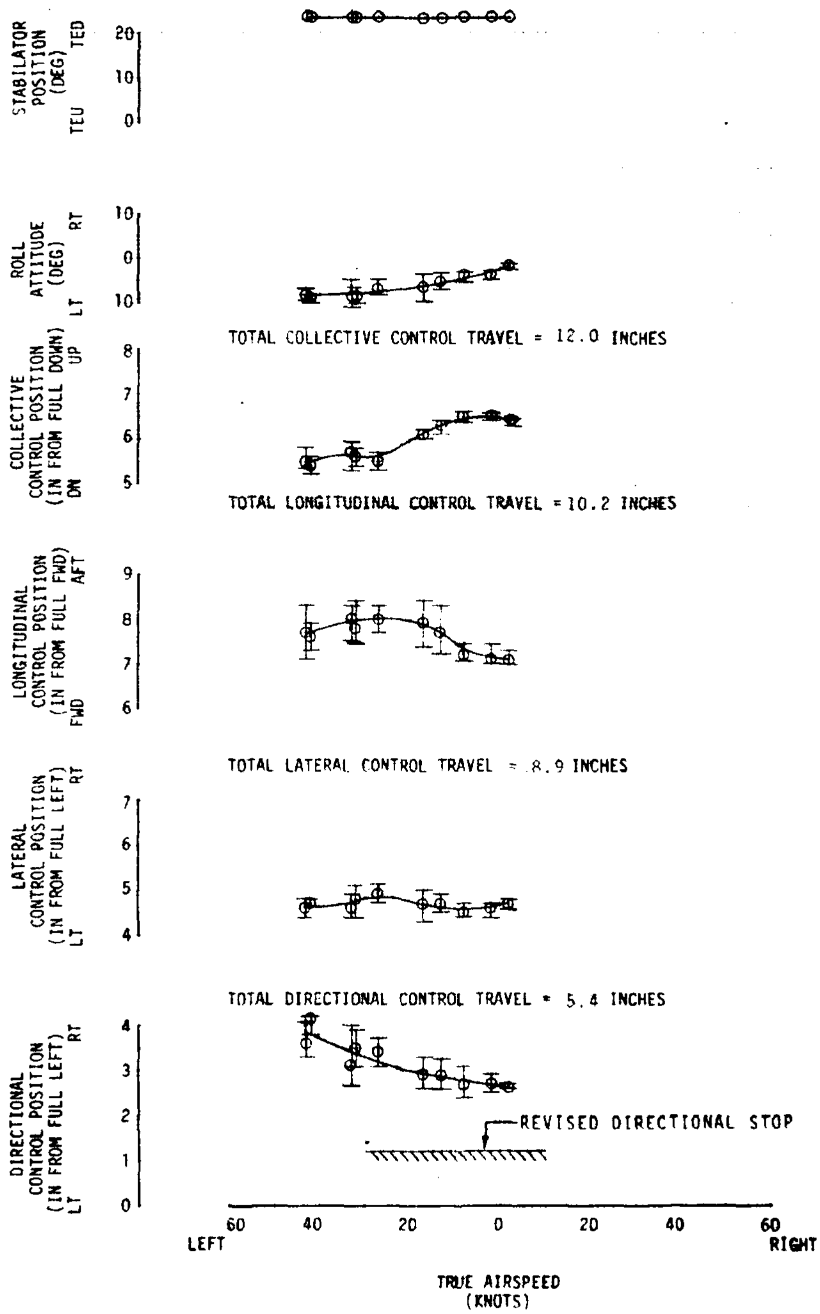


FIGURE 37
DIRECTIONAL CONTROL INPUT
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	WHEEL HEIGHT (FT)
14400	201.9(FWD)	0.0	-194	11.0	290	20

NOTES: 1. 8-HELLFIRE CONFIGURATION
2. INPUT MADE AT 35 KTAS RIGHT SIDEWARD FLIGHT
3. DASE ON

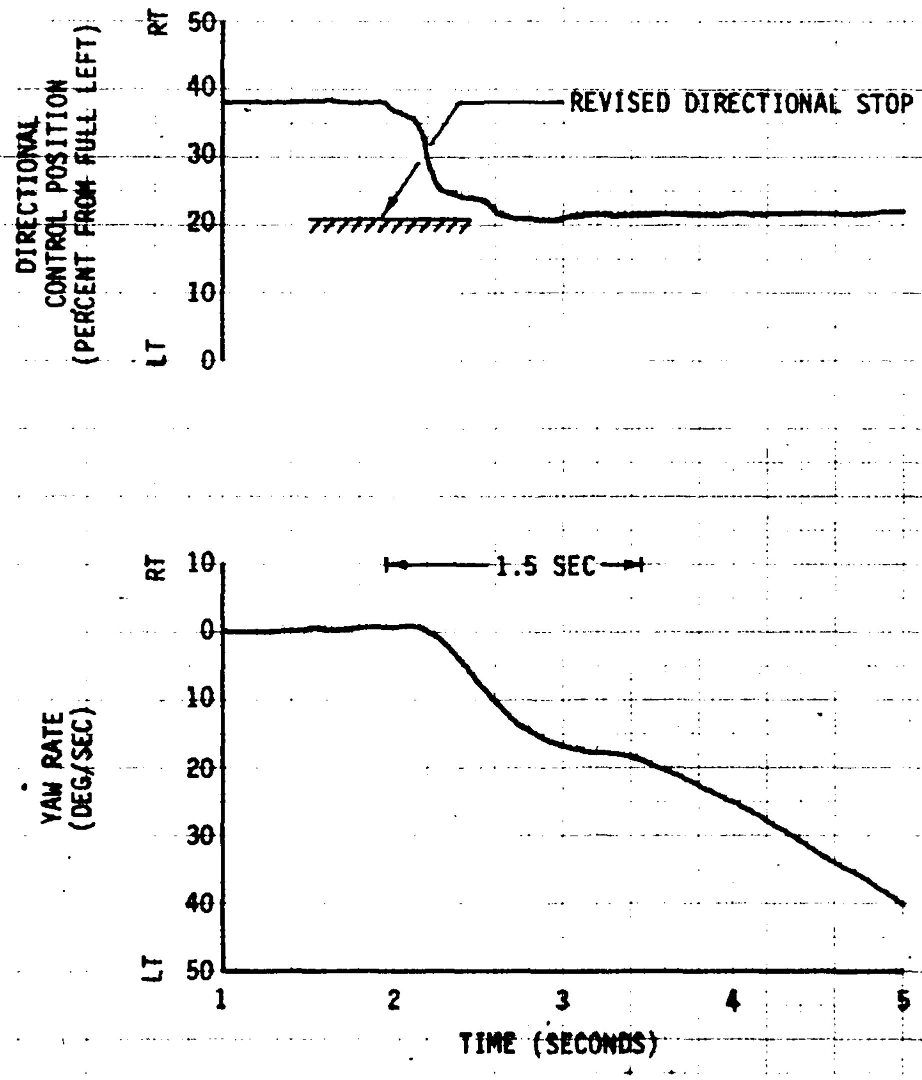


FIGURE 38
LATERAL REVERSAL
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	TRIM ROTOR SPEED (RPM)	DASE CONDITION	CONFIGURATION
14,700	201.9(AFT)	0.0	20	12.5	290	ON	8-HELLFIRE

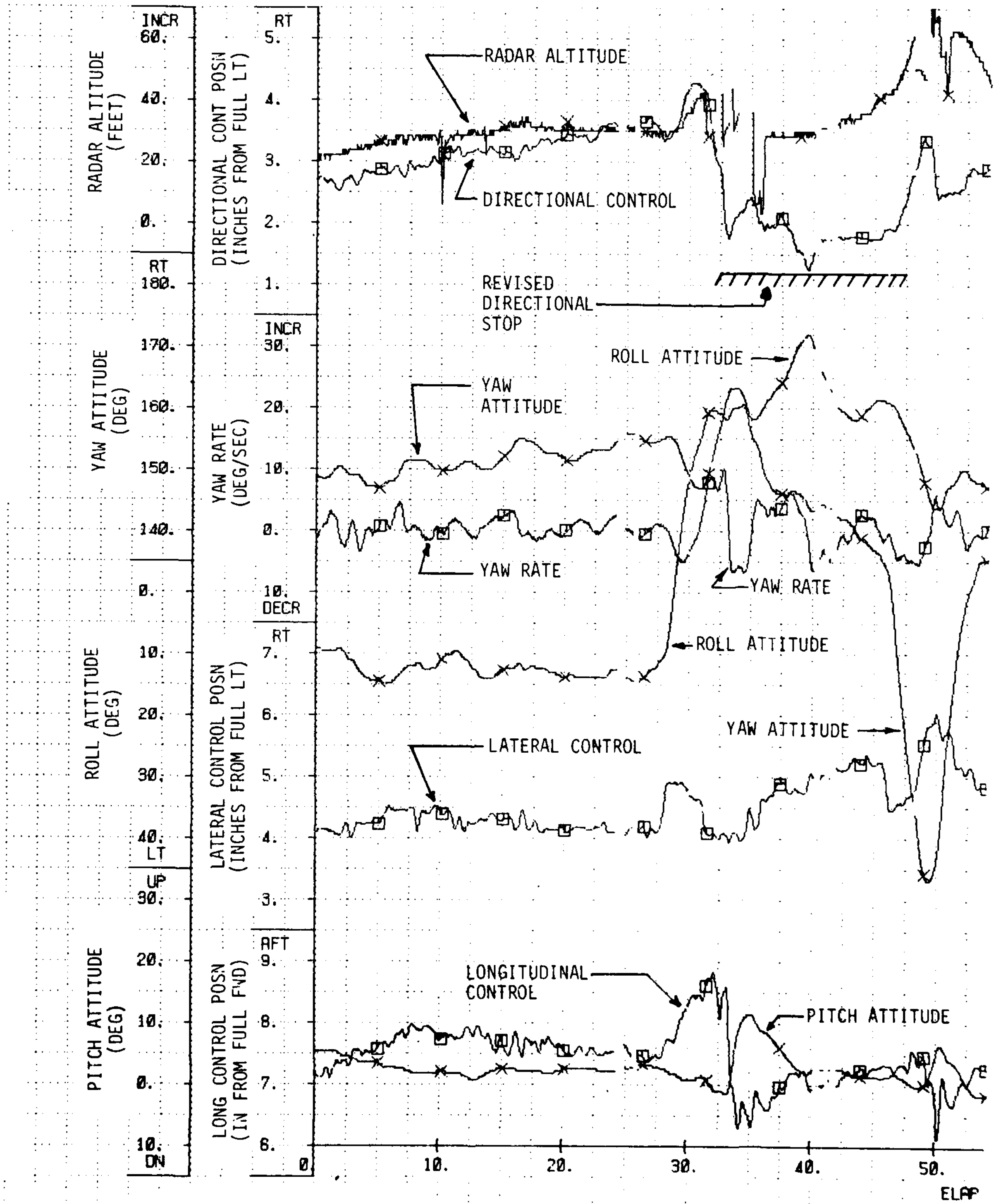


FIGURE 39
SINGLE ENGINE FAILURE
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	DASE CONDITION	FLIGHT CONDITION
14480	205.7(AFT)	0.0	3740	11.0	289	76	OFF	IRP CLIMB

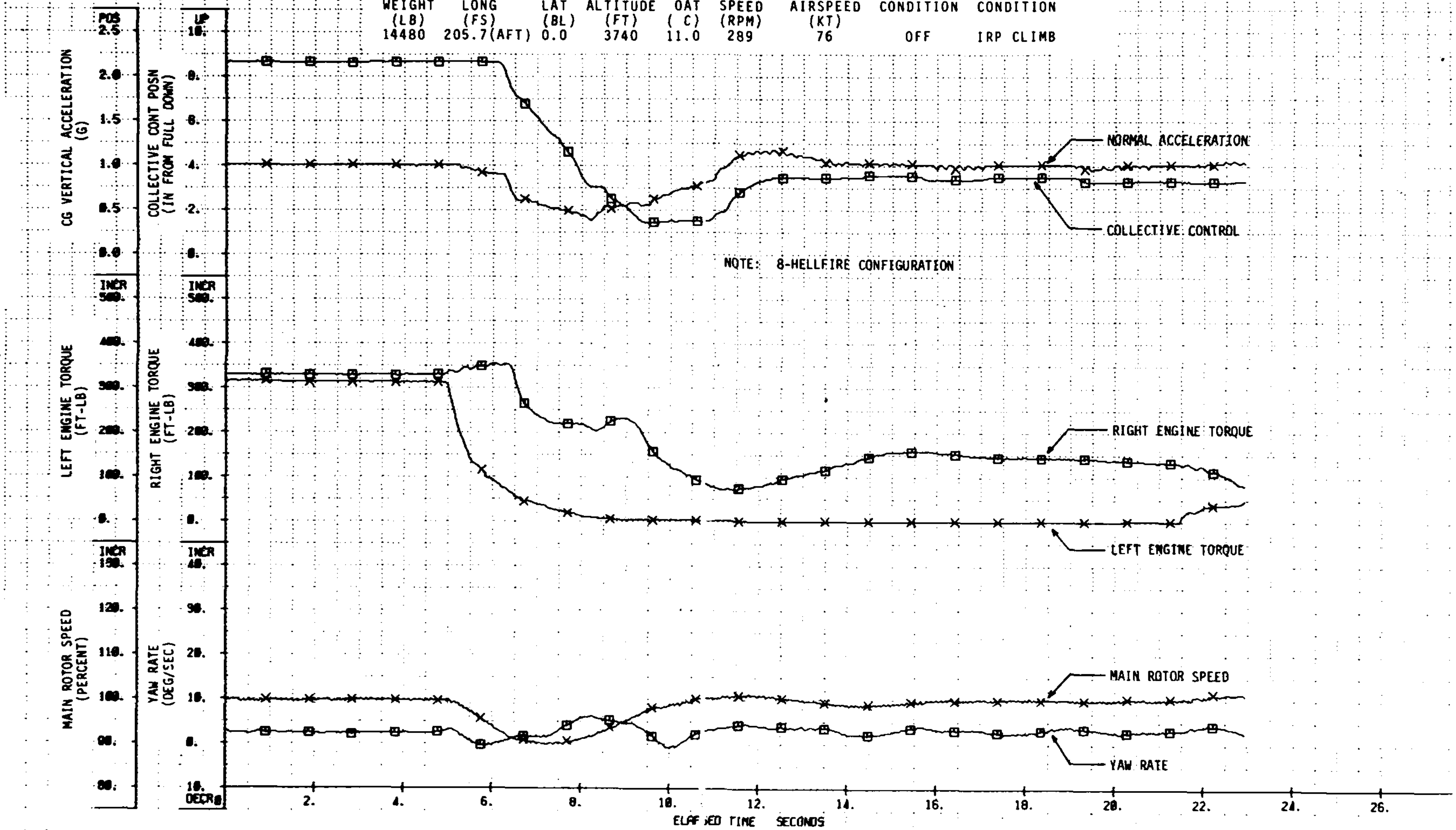


FIGURE 40
SINGLE ENGINE FAILURE
YAH-64 USA S/N 77-23258

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	DASE CONDITION	CONFIGURATION
14360	206.7(AFT)	0.0	4640	8.0	289	144	ON	8-HELLFIRE

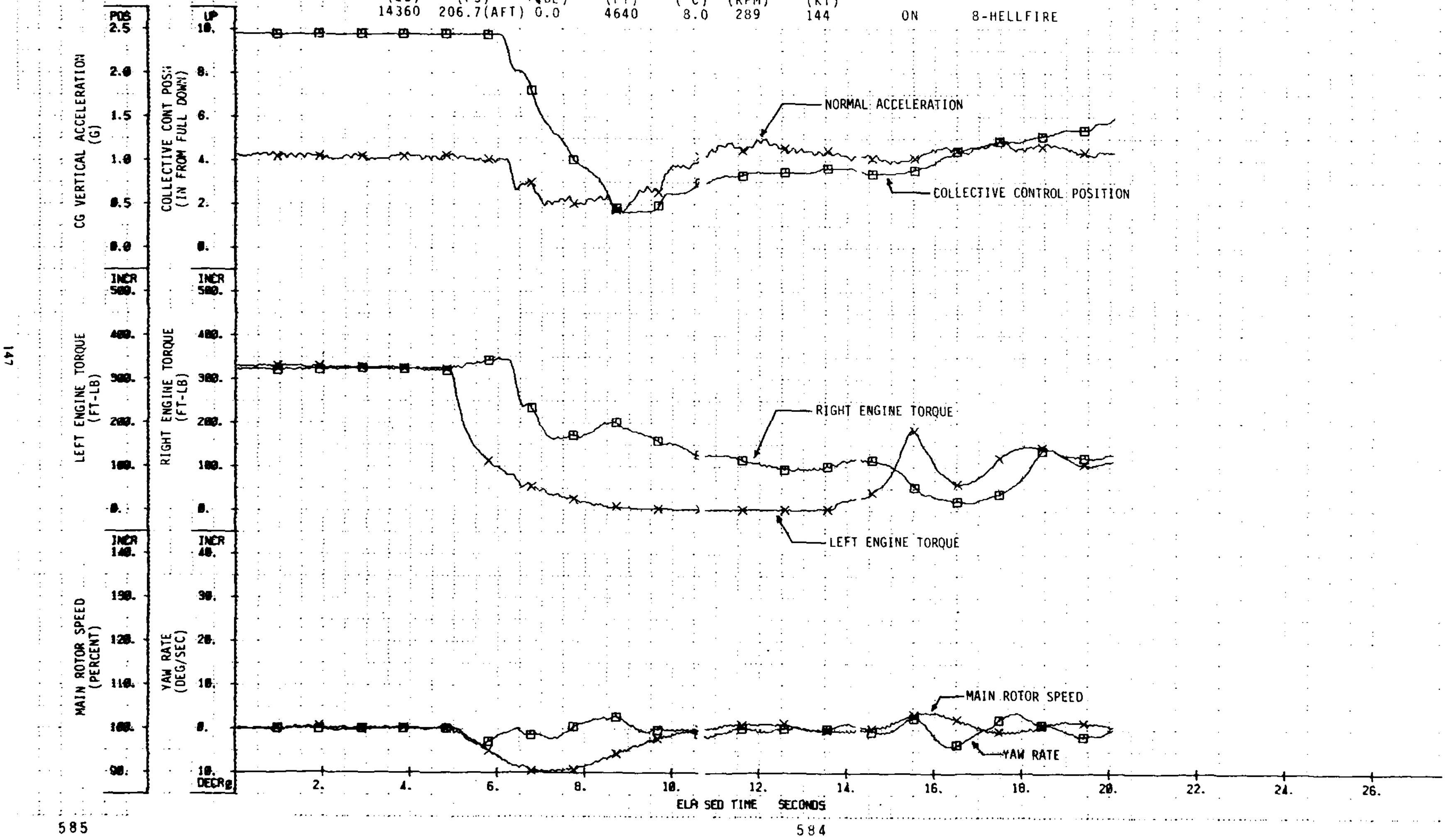
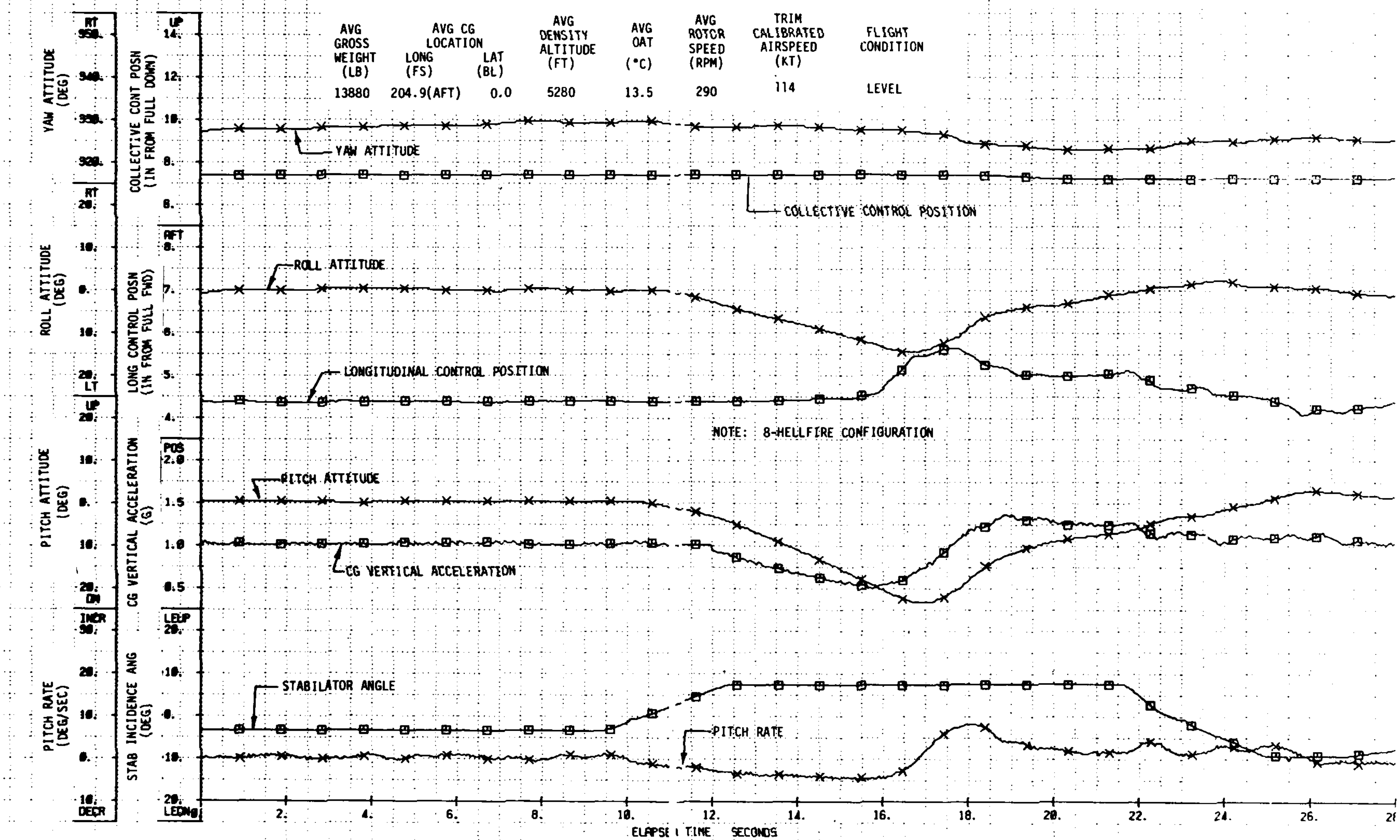


FIGURE 41
STABILATOR FAILURE
YAH-64 USA S/N 77-23258



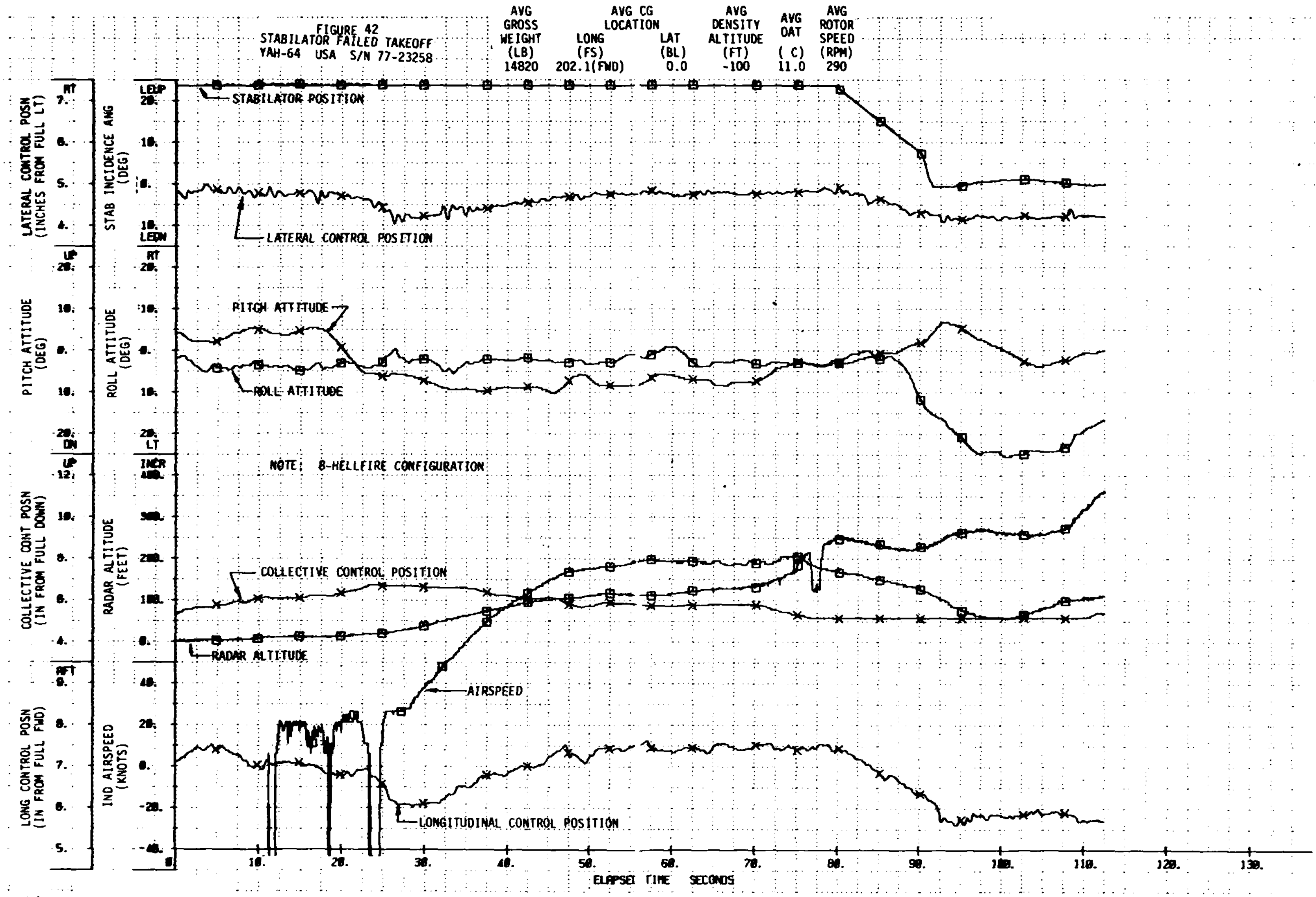


FIGURE 42
STABILATOR FAILED TAKEOFF
YAH-64 USA S/N 77-23258

264

263

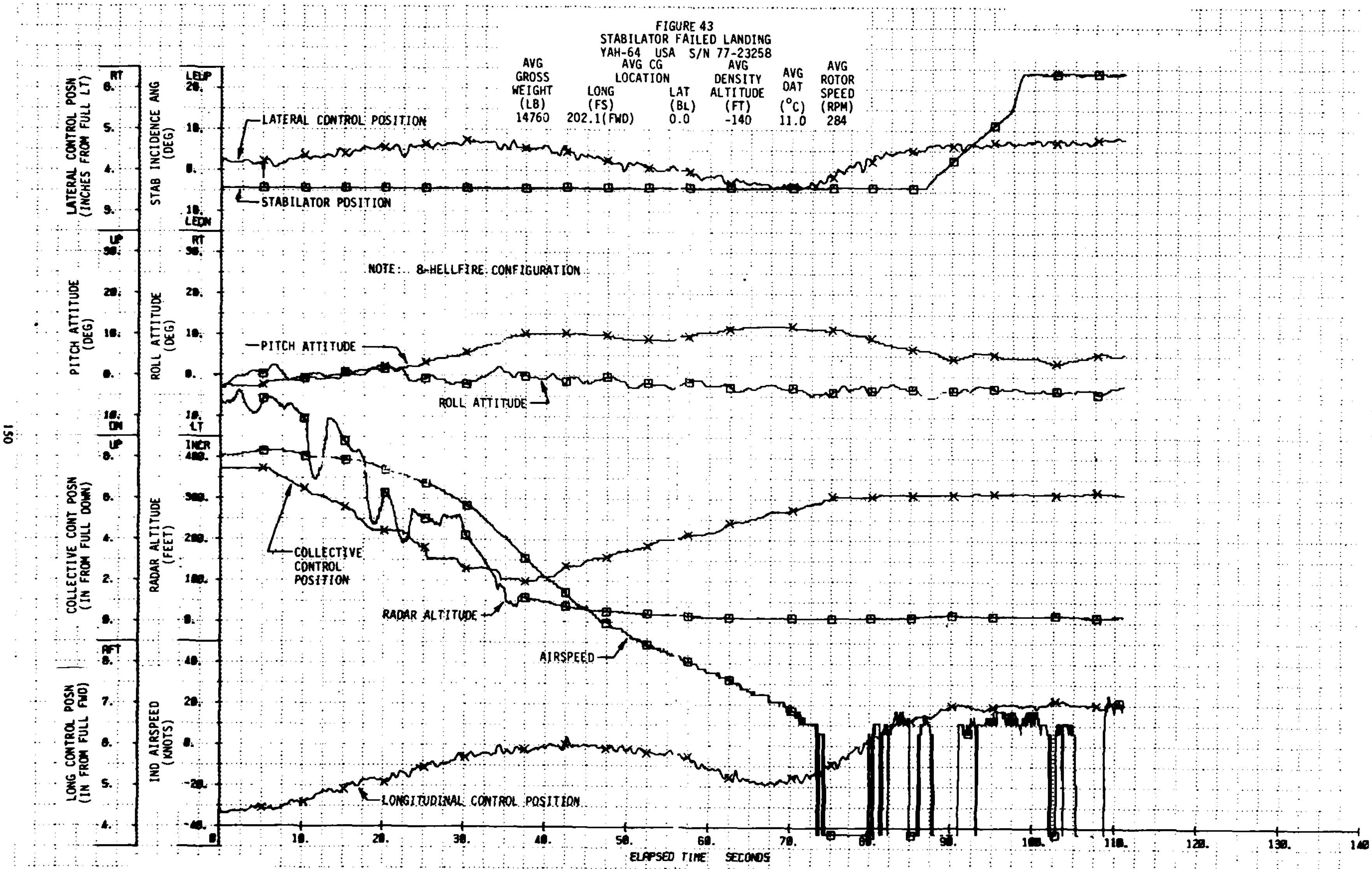


FIGURE 43
STABILATOR FAILED LANDING
YAH-64 USA S/N 77-23258

260

259

FIGURE 44
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258
PILOT SEAT

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
14720	201.4(FWD)	0.0	280	14.0	289	LOWSPEED

NOTE 1 8 HELLFIRE CONFIGURATION
2 WHEEL HEIGHT 20 FEET

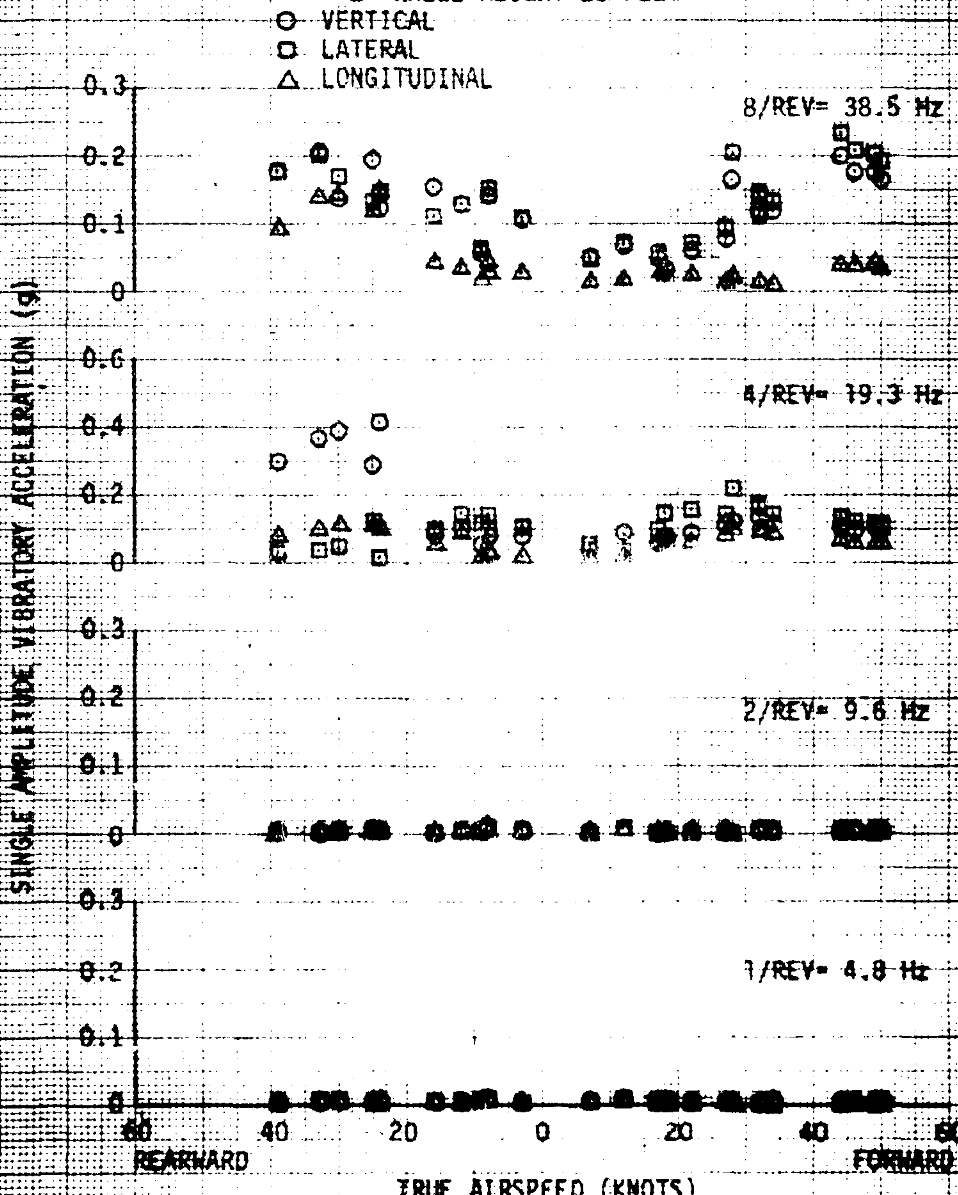
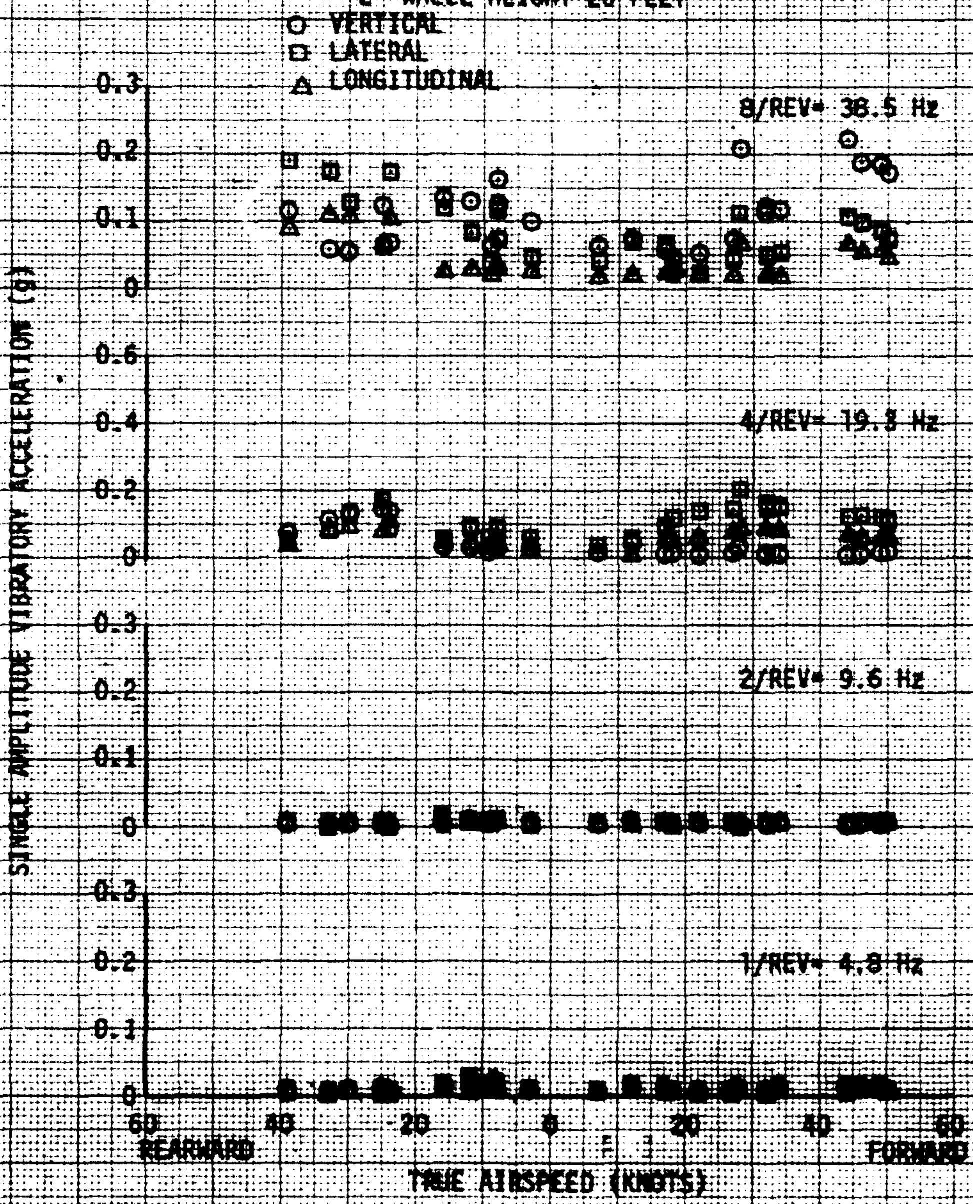


FIGURE 45
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258
CO PILOT SEAT

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
14720	201.4(FWD)	0.0	280	14.0	289	LOWSPEED

NOTE 1 8 HELLFIRE CONFIGURATION
2 WHEEL HEIGHT 20 FEET



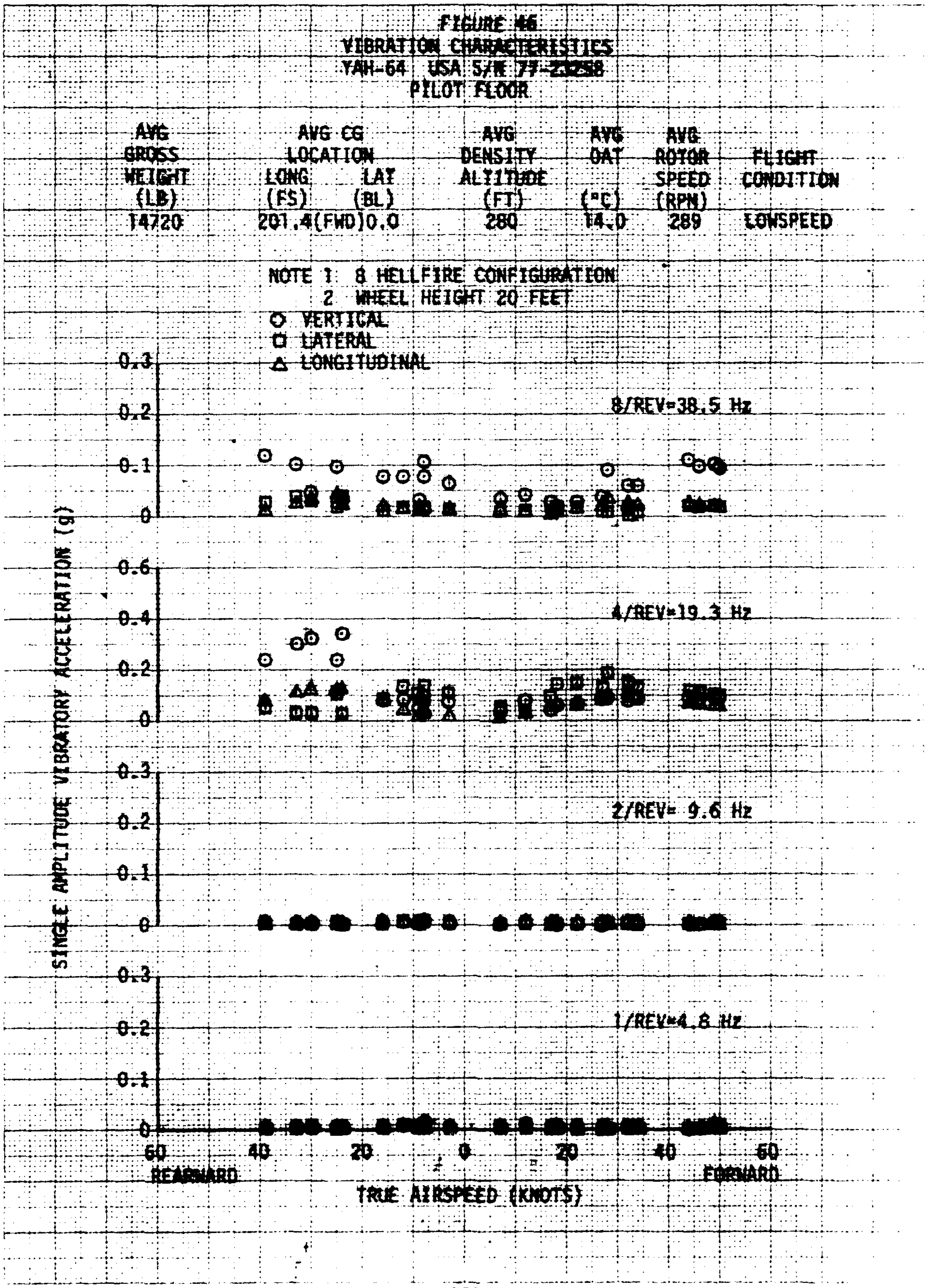


FIGURE 46
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258
PILOT FLOOR
AVG GROSS WEIGHT (LB) 14720
AVG CG LOCATION LONG (FS) 201.4(FWD) LAT (BL) 0.0
AVG DENSITY ALTITUDE (FT) 280
AVG OAT (°C) 14.0
AVG ROTOR SPEED (RPM) 289
FLIGHT CONDITION LOWSPEED
NOTE 1 8 HELLFIRE CONFIGURATION
2 WHEEL HEIGHT 20 FEET
VERTICAL
LATERAL
LONGITUDINAL
8/REV=38.5 Hz
4/REV=19.3 Hz
2/REV= 9.6 Hz
1/REV=4.8 Hz
SINGLE AMPLITUDE VIBRATORY ACCELERATION (g)
REARWARD
FORWARD
TRUE AIRSPEED (KNOTS)

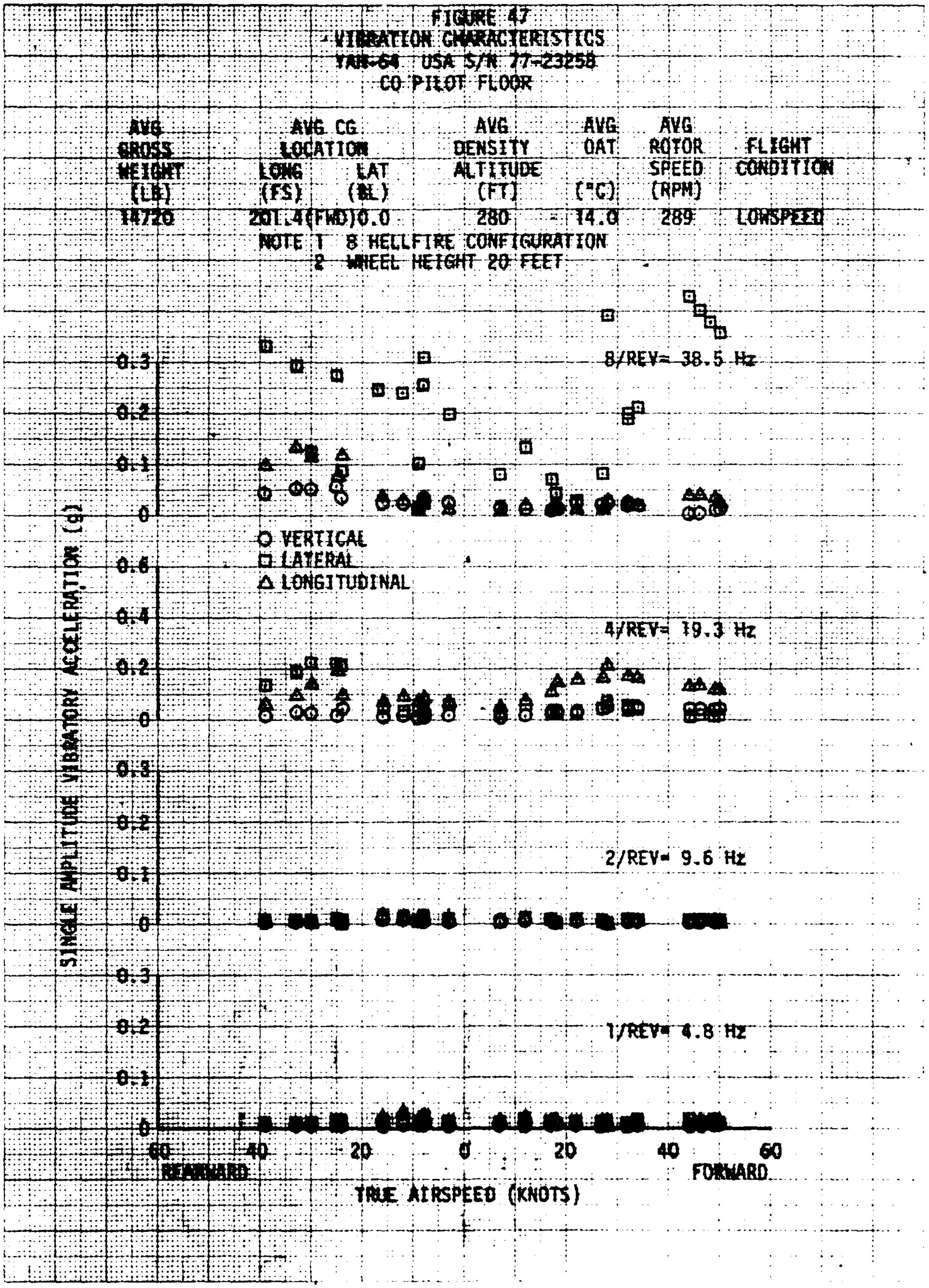


FIGURE 47
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258
CO PILOT FLOOR
AVG GROSS WEIGHT (LB)
14720
AVG CG LOCATION
LONG (FS)
201.4(FWD)
LAT (BL)
0.0
AVG DENSITY ALTITUDE (FT)
280
AVG OAT (°C)
14.0
AVG ROTOR SPEED (RPM)
289
FLIGHT CONDITION
LOWSPEED
NOTE 1 8 HELLFIRE CONFIGURATION
2 WHEEL HEIGHT 20 FEET
8/REV= 38.5 Hz
4/REV= 19.3 Hz
2/REV= 9.6 Hz
1/REV= 4.8 Hz
O VERTICAL
□ LATERAL
△ LONGITUDINAL
SINGLE AMPLITUDE VIBRATORY ACCELERATION (g)
0.3
0.2
0.1
0
0.6
0.4
0.2
0
0.3
0.2
0.1
0
0.3
0.2
0.1
0
60
40
20
0
20
40
60
REARWARD
FORWARD
TRUE AIRSPEED (KNOTS)

FIGURE 48
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258
AIRCRAFT CG

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
14720	201.4(FWD)	0.0	280	14.0	289	LOWSPEED

NOTE 1 8 HELLFIRE CONFIGURATION
2 WHEEL HEIGHT 20 FEET

○ VERTICAL
□ LATERAL
△ LONGITUDINAL

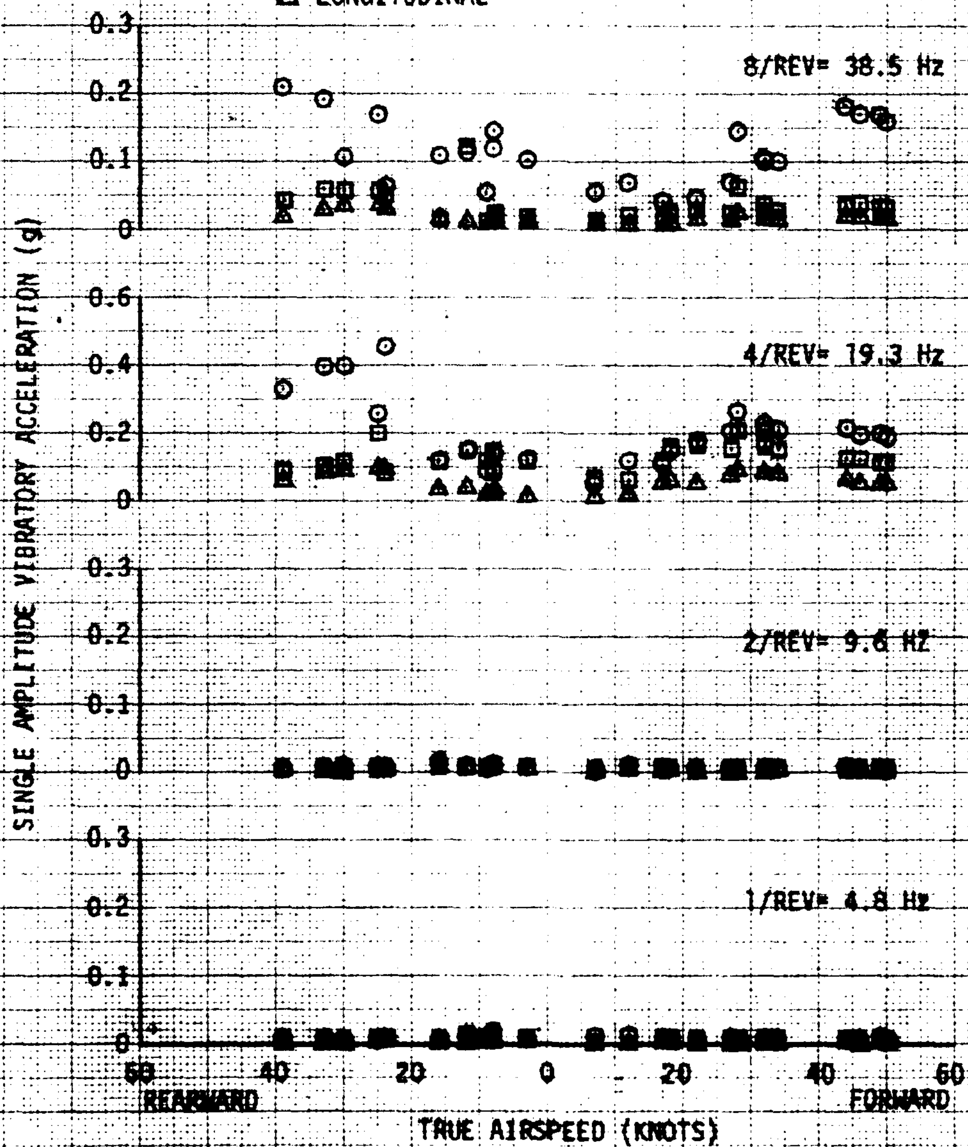


FIGURE 49
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258
PILOT SEAT

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
14820	201.3(FWD)	0.0	440	15	290	LOWSPEED

NOTE 1 8 HELLFIRE CONFIGURATION
2 WHEEL HEIGHT 20 FEET

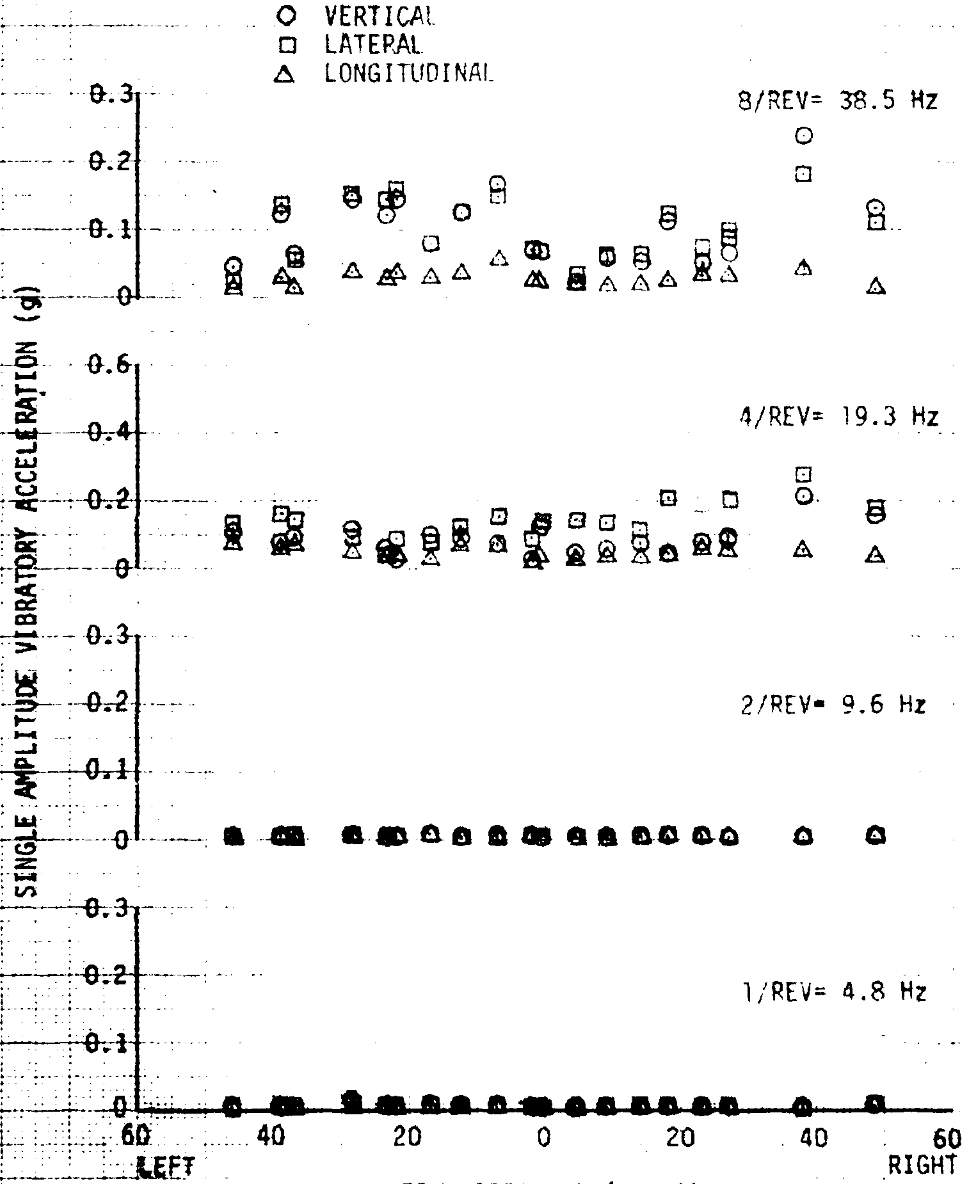


FIGURE 50
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258
CO PILOT SEAT

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
14820	201.3(FWD)	0.0	440	15	290	LOWSPEED

NOTE 1 8 HELLFIRE CONFIGURATION
2 WHEEL HEIGHT 20 FEET

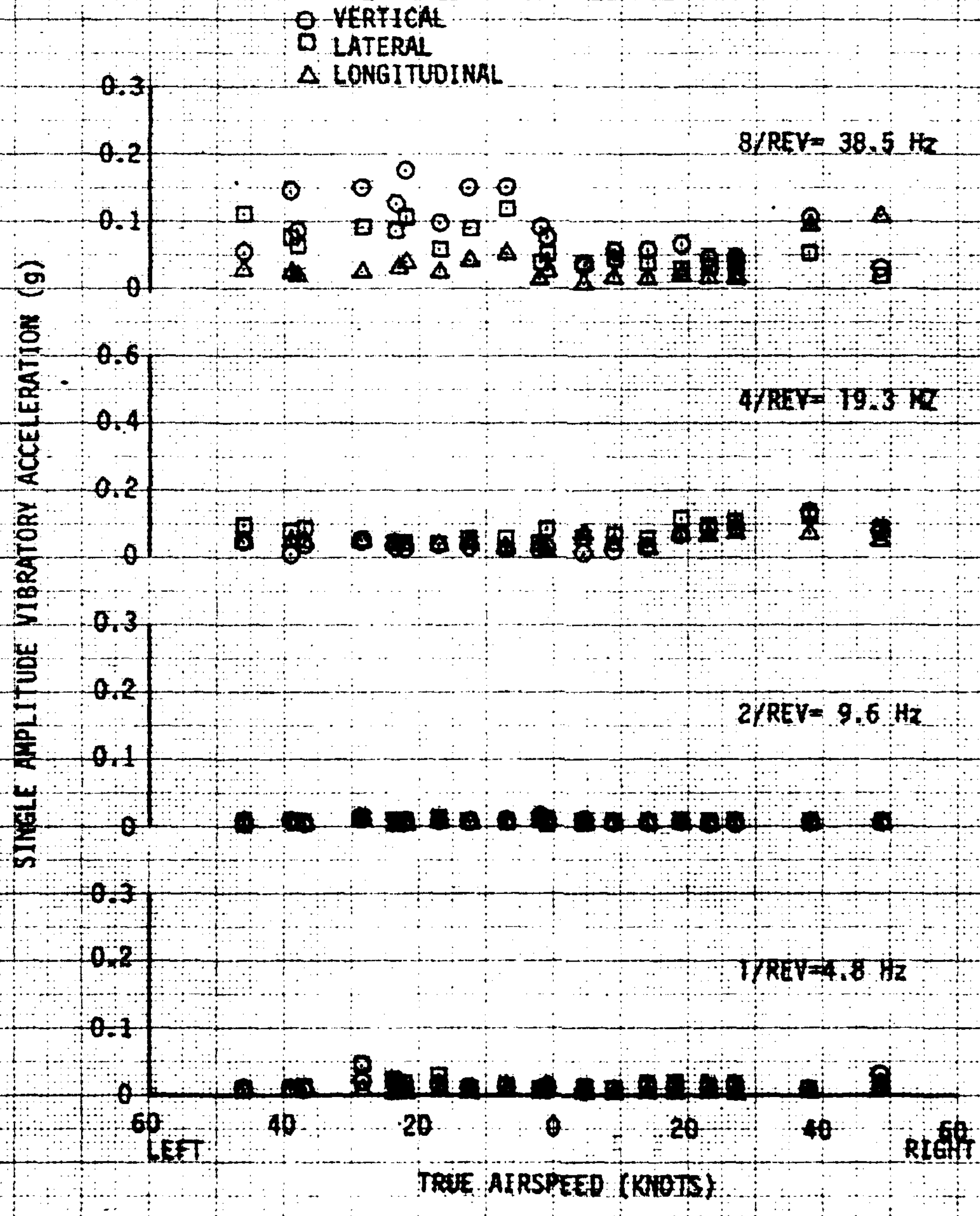


FIGURE 51
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258
PILOT FLOOR

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
14820	201.3(FWD)	0.0	440	15	290	LOWSPEED

NOTE 1 8 HELLFIRE CONFIGURATION
2 WHEEL HEIGHT 20 FEET

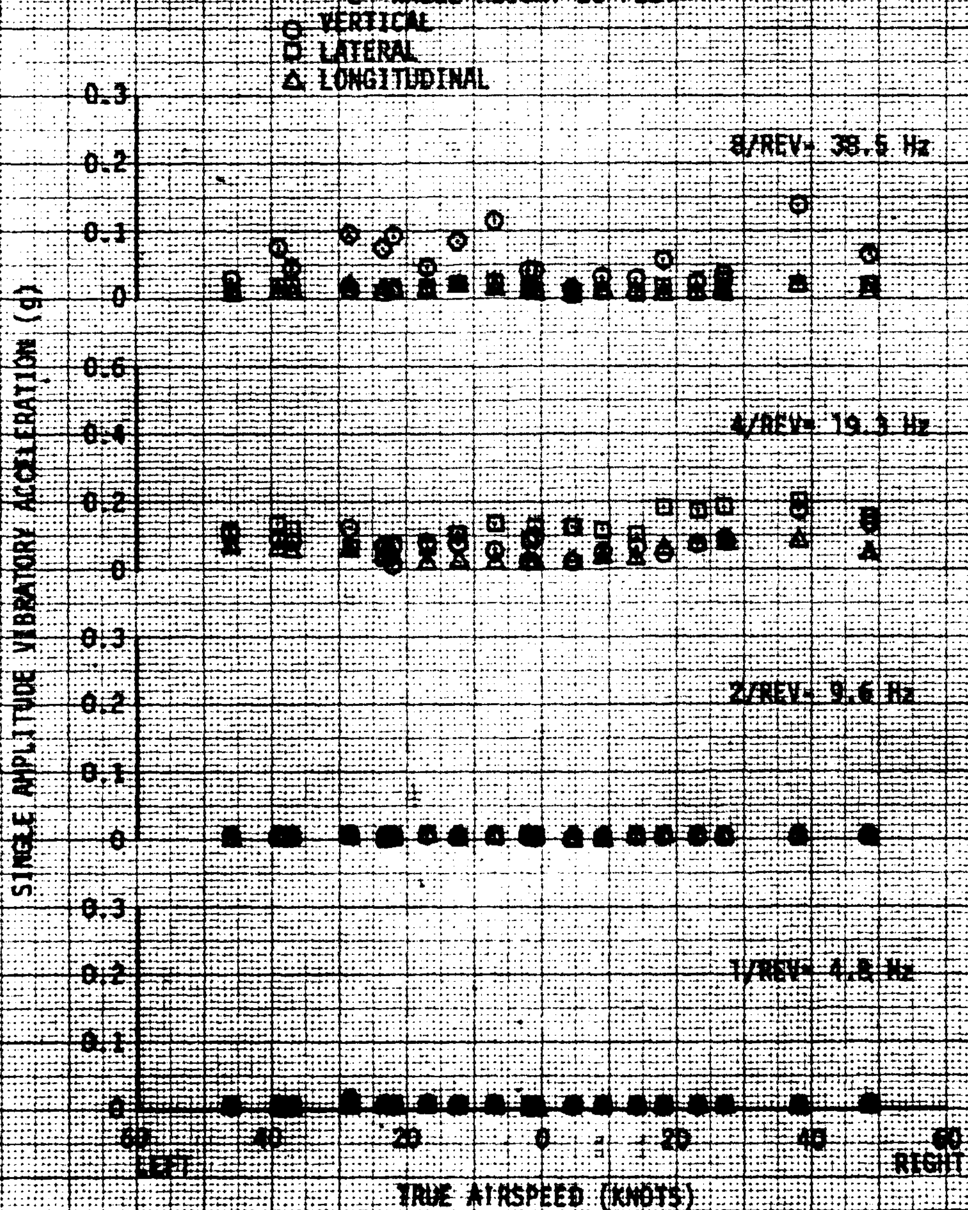


FIGURE 52
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258
CO PILOT FLOOR

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
14820	201.3(FWD)	0.0	440	15	290	LOWSPEED

NOTE 1 8 HELLFIRE CONFIGURATION
2 WHEEL HEIGHT 20 FEET

○ VERTICAL
□ LATERAL
△ LONGITUDINAL

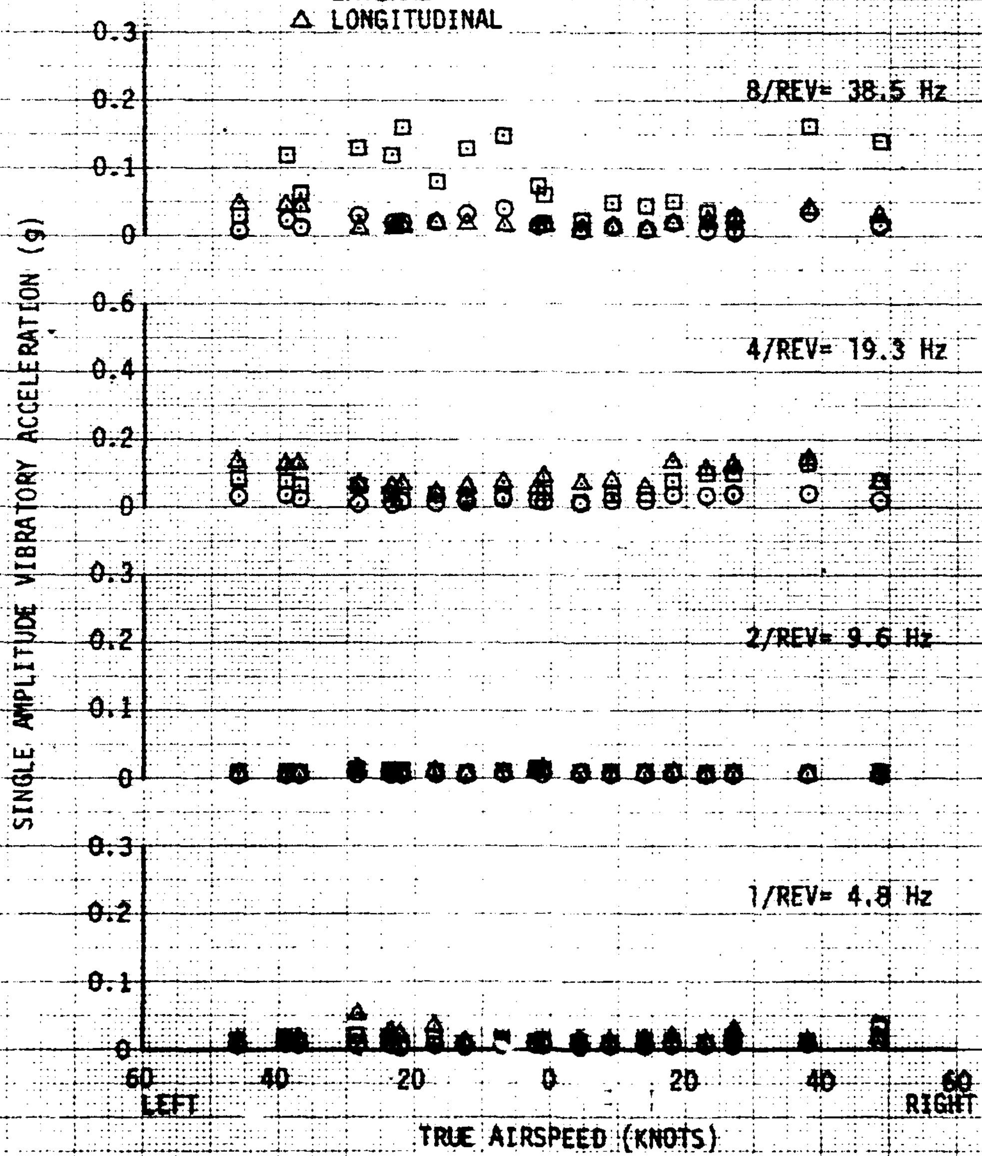


FIGURE 53
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258
AIRCRAFT CG

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
14820	201.3(FWD)	0.0	440	15	290	LOWSPEED

NOTE 1 8 HELLFIRE CONFIGURATION
2 WHEEL HEIGHT 20 FEET

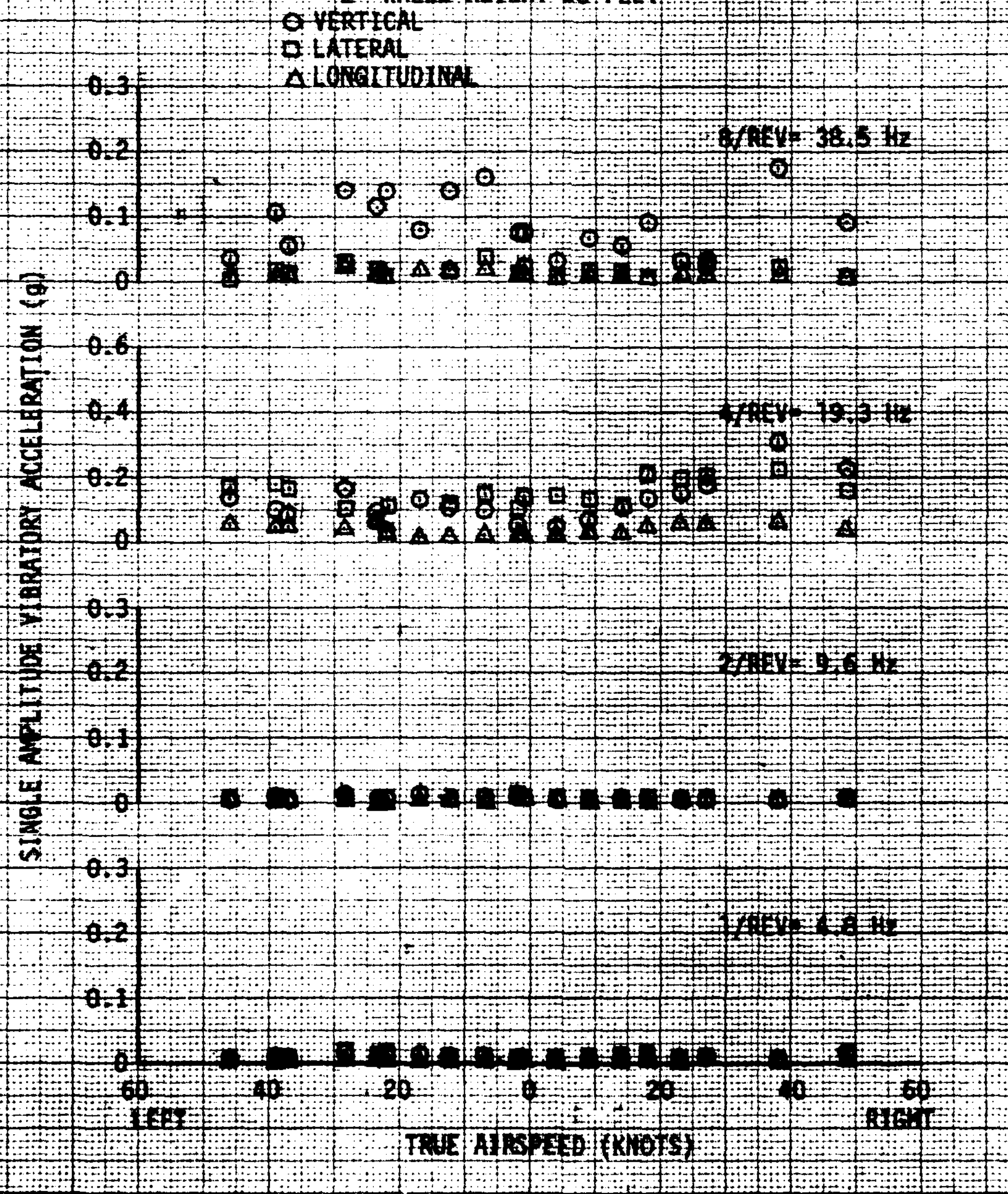


FIGURE 54
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258
PILOTS SEAT

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
15600	202.3(FWD)	0.0	5700	8.0	289	CLIMB

NOTE 8 HELLFIRE CONFIGURATION
O VERTICAL
□ LATERAL
△ LONGITUDINAL

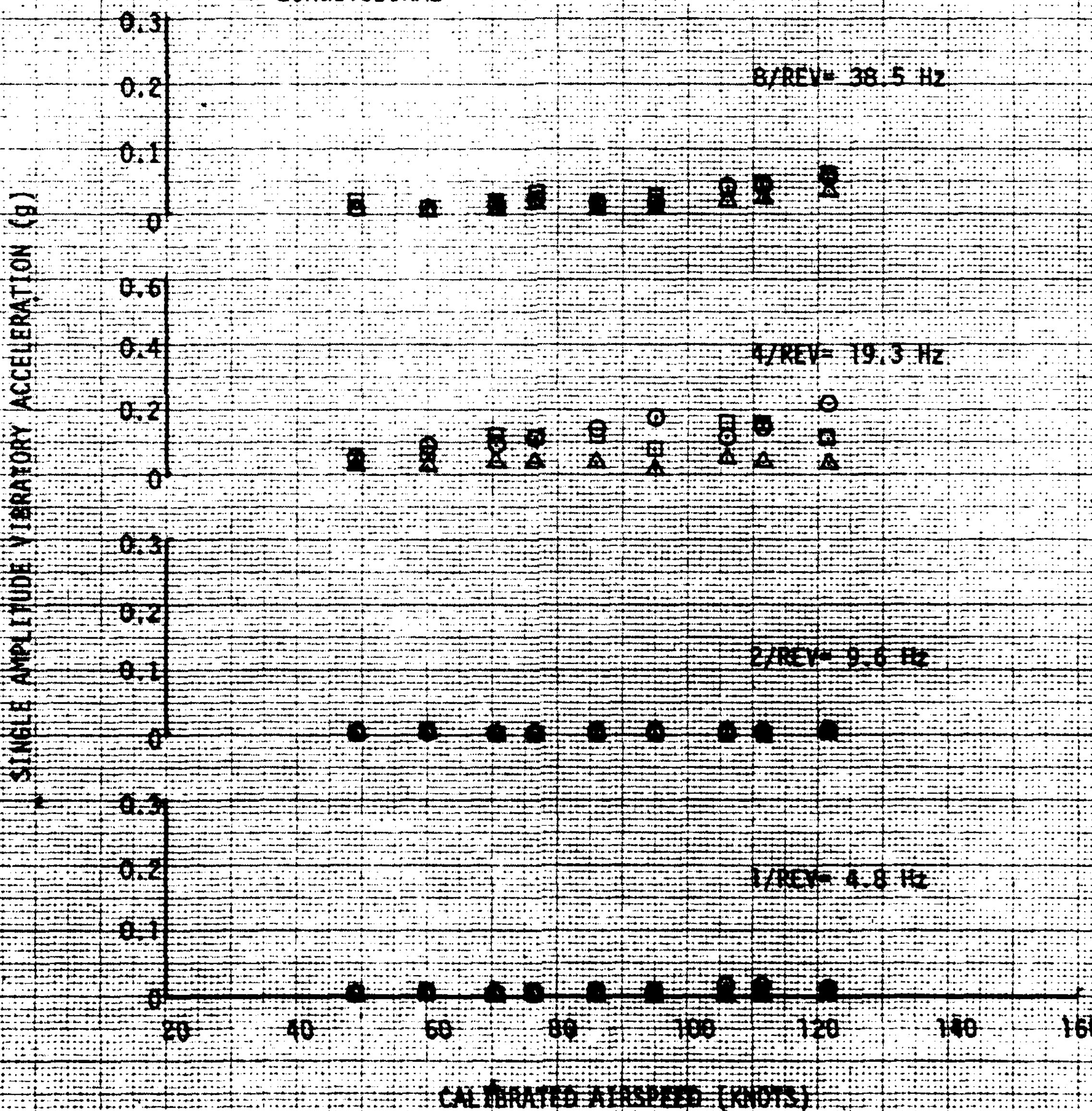


FIGURE 55
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258
CO PILOT SEAT

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
15600	202.3(FWD)	0.0	5700	8.0	289	CLIMB

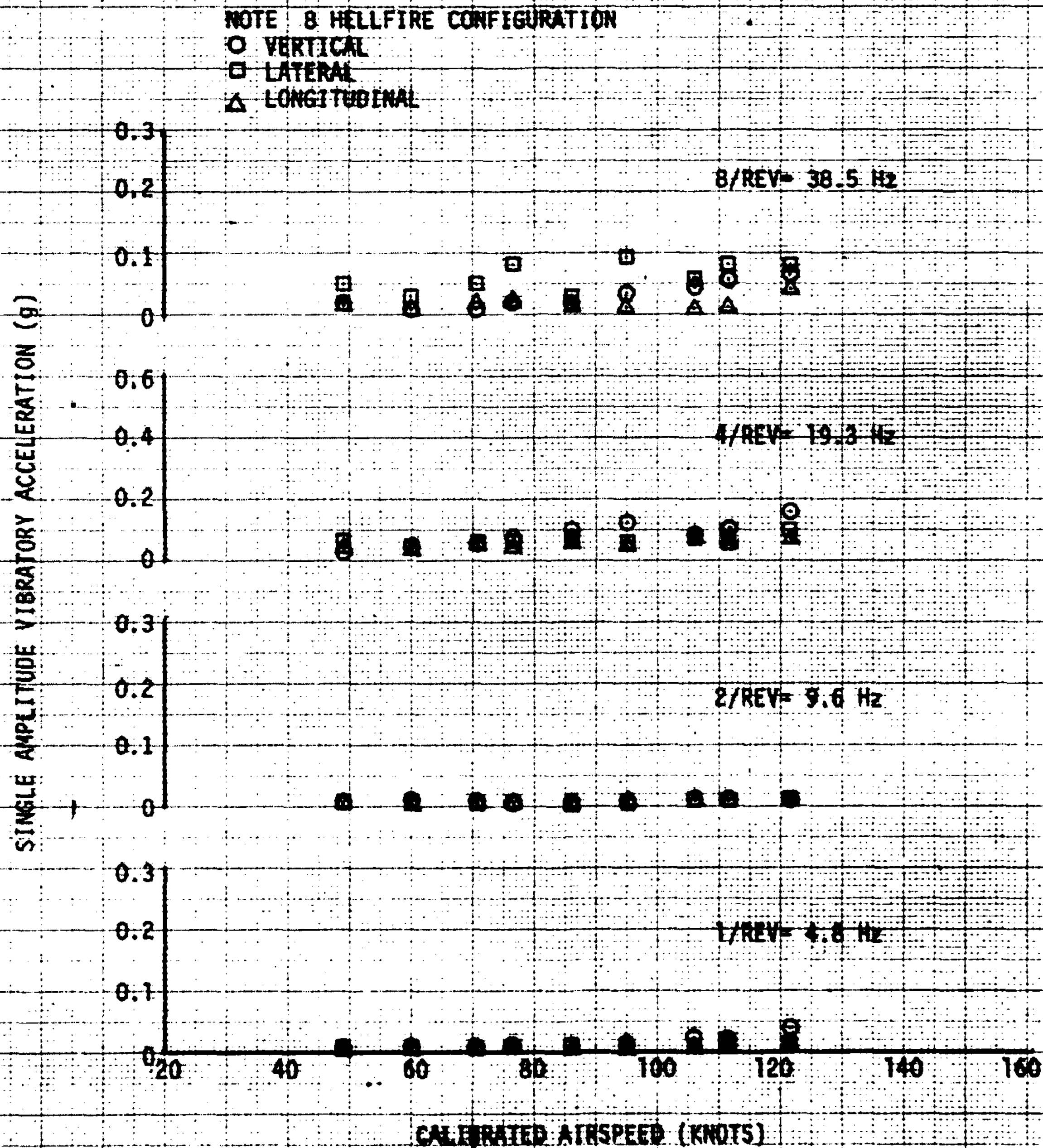


FIGURE 56
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258
AIRCRAFT CG

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
15600	202.3(FWD)	0.0	5700	8.0	289	CLIMB

NOTE: 8-HELLFIRE CONFIGURATION

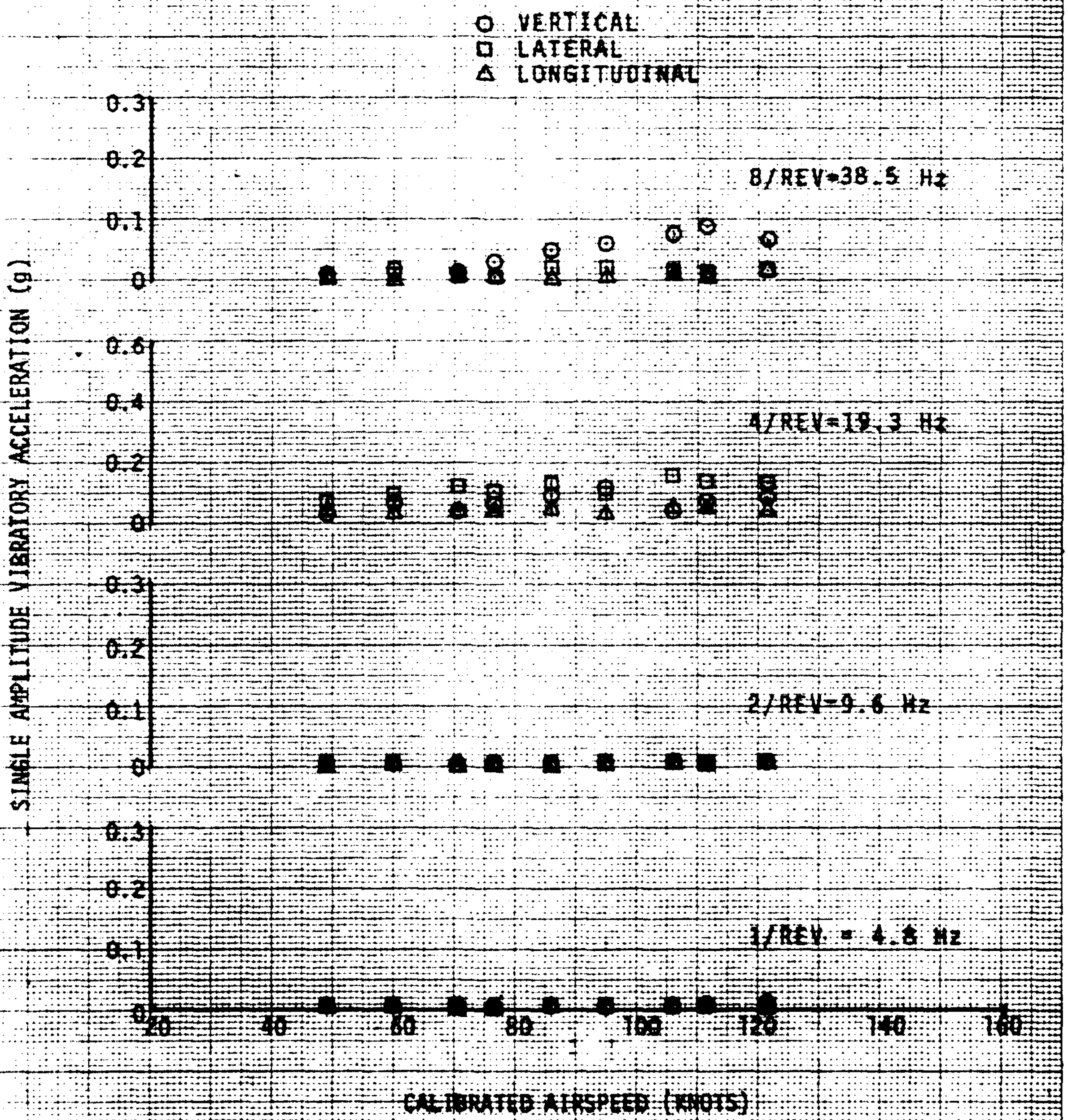


FIGURE 57
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258
PILOTS SEAT

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
15580	202.3(FWD)	0.0	4480	9.0	290	DESCENT

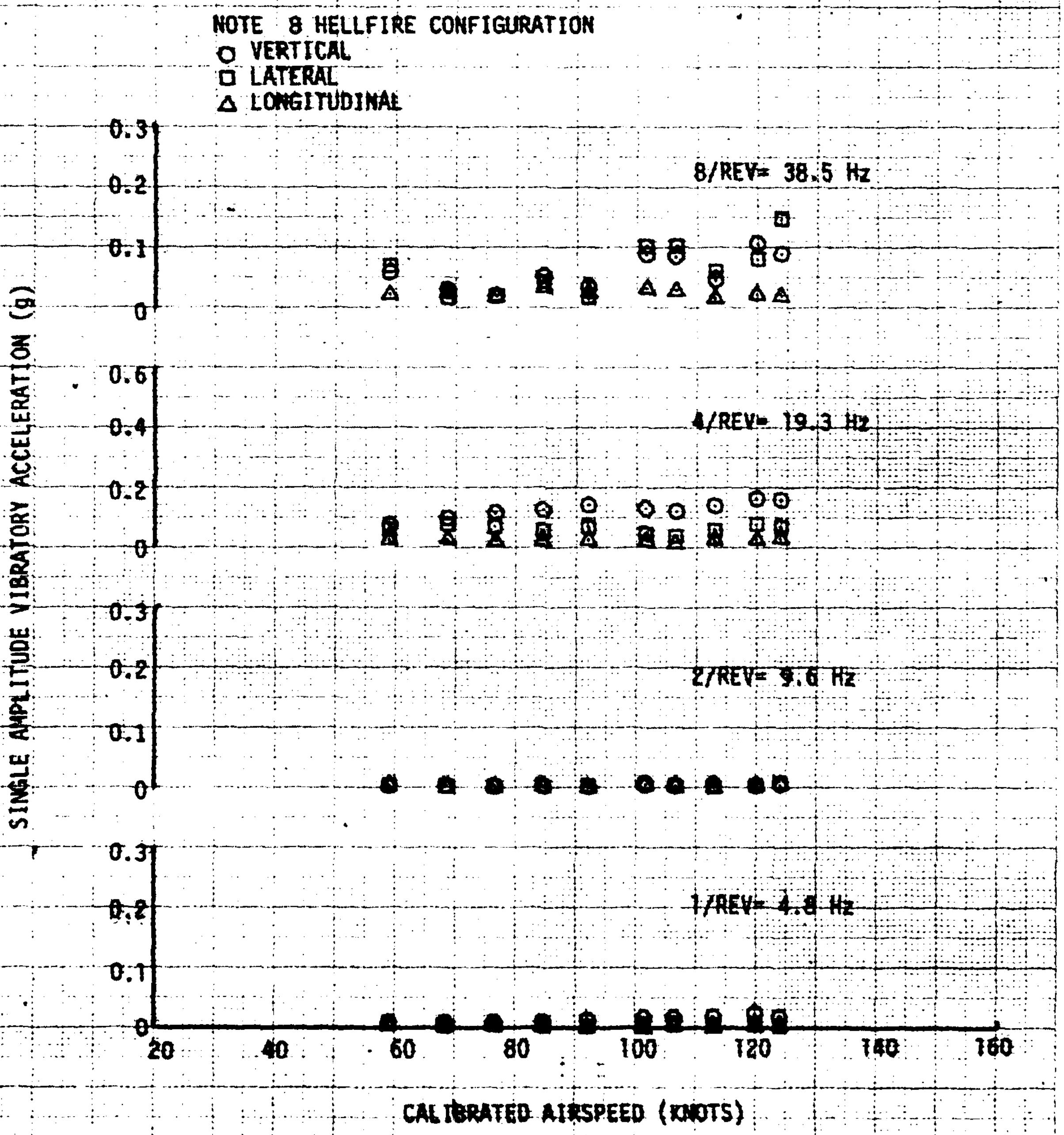


FIGURE 58
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258
CO PILOT SEAT

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
15580	202.3(FWD)	0.0	4480	9.0	290	DESCENT

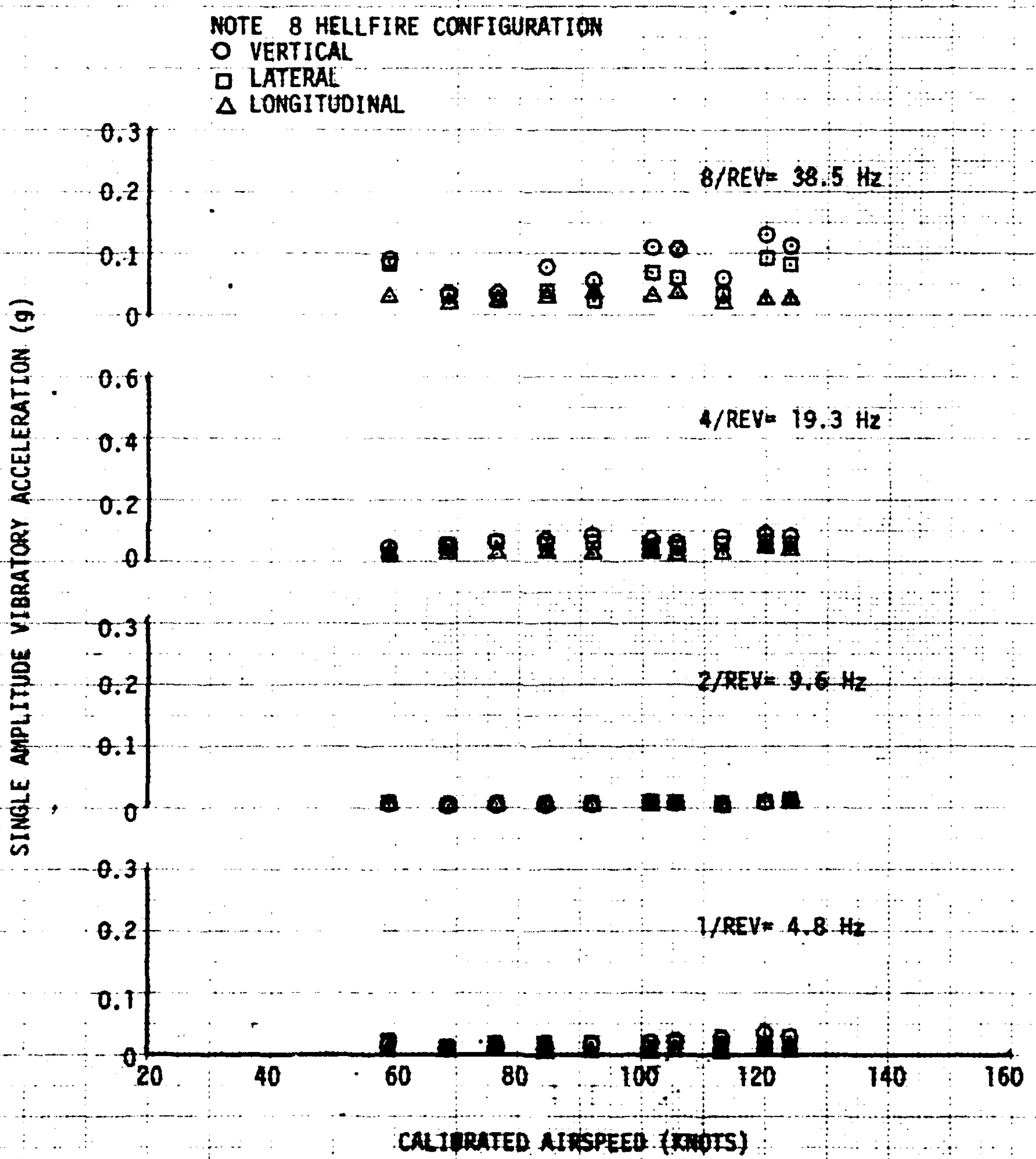


FIGURE 59
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258
AIRCRAFT CG

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
15580	202.3(FWD)	0.0	4480	9.0	290	DESCENT

NOTE: 8-HELLFIRE CONFIGURATION

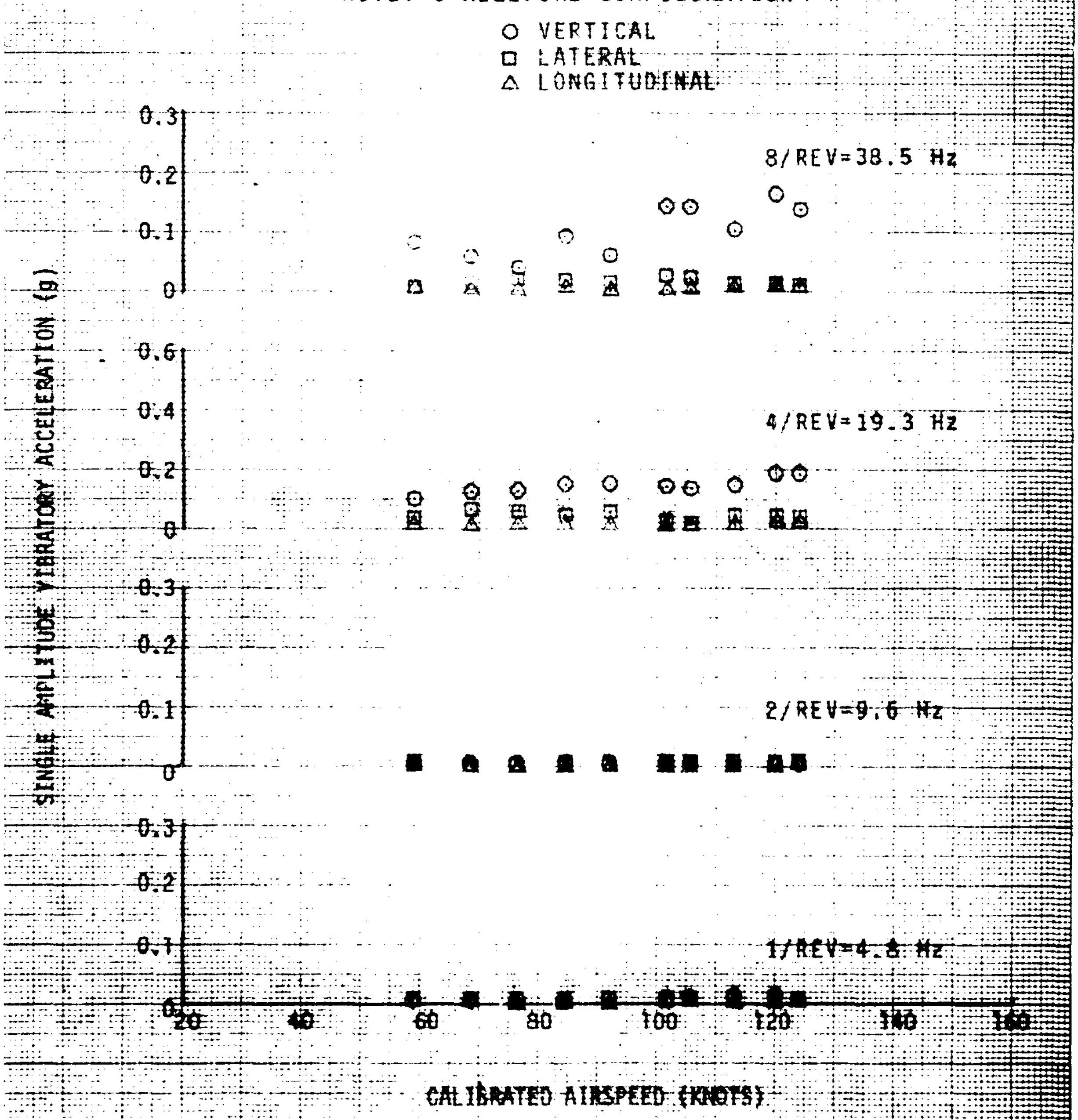


FIGURE 60
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258
PILOT SEAT

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
14600	202.0(FWD)	0.0	1360	17.5	290	LEVEL

NOTE: 8-HELLFIRE CONFIGURATION

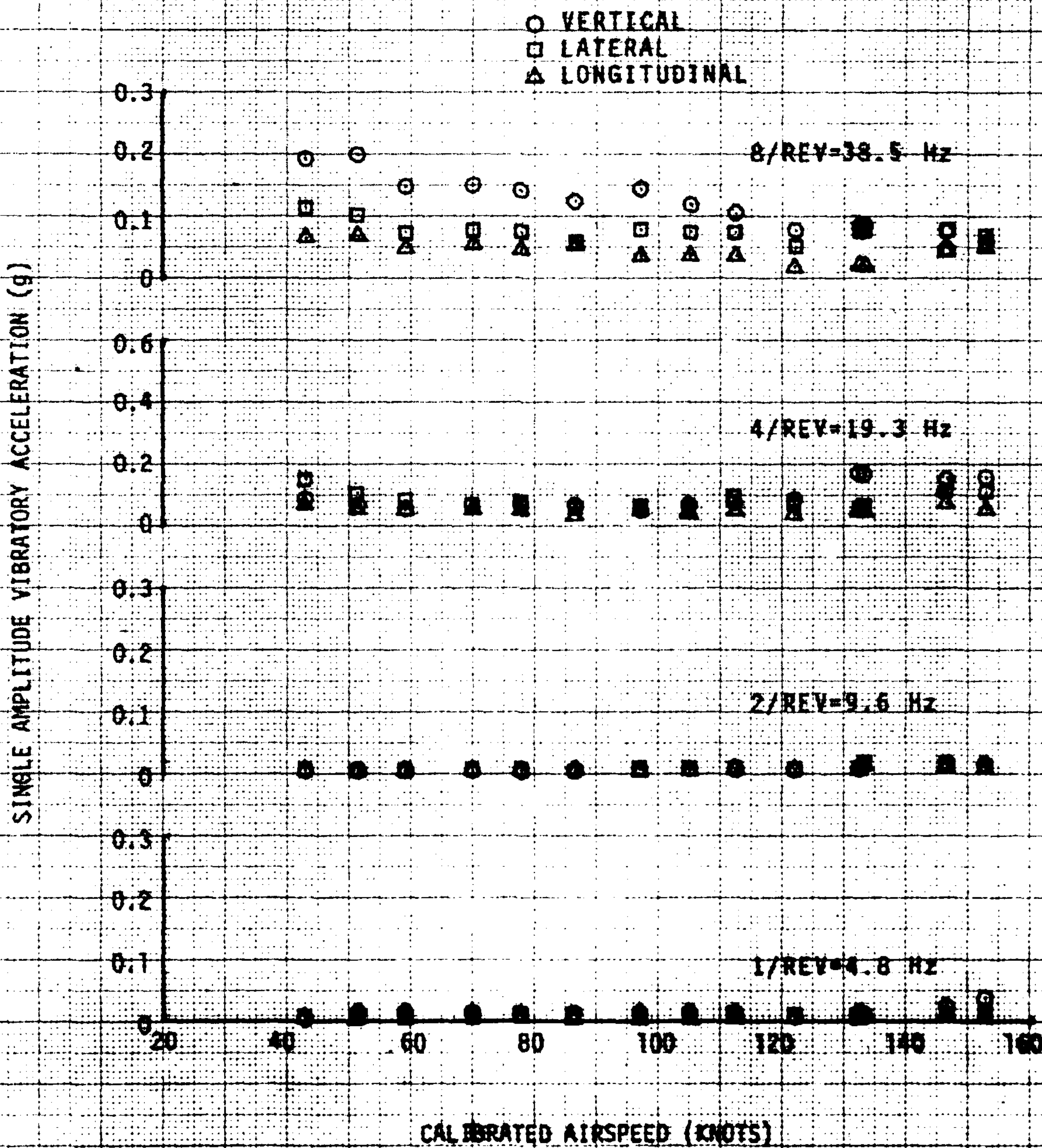


FIGURE 61
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258
COPILOT SEAT

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
14600	202.0(FWD)	0.0	1360	17.5	289	LEVEL

NOTE: 8-HELLFIRE CONFIGURATION

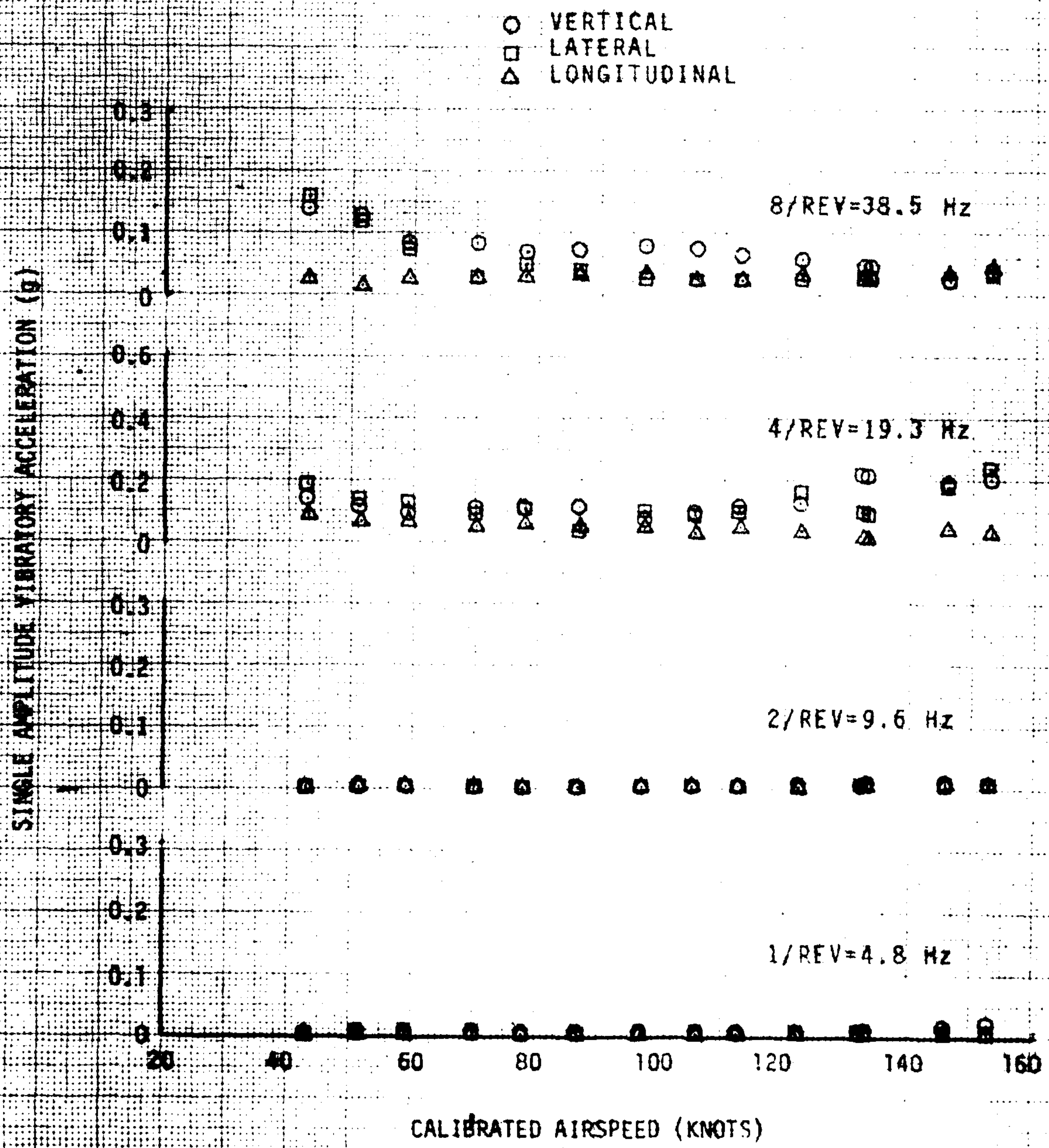


FIGURE 62
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258
PILOT SEAT

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
14860	202.0(FWD)	0.0	5420	10.0	290	LEVEL

NOTE: 8 HELLFIRE CONFIGURATION

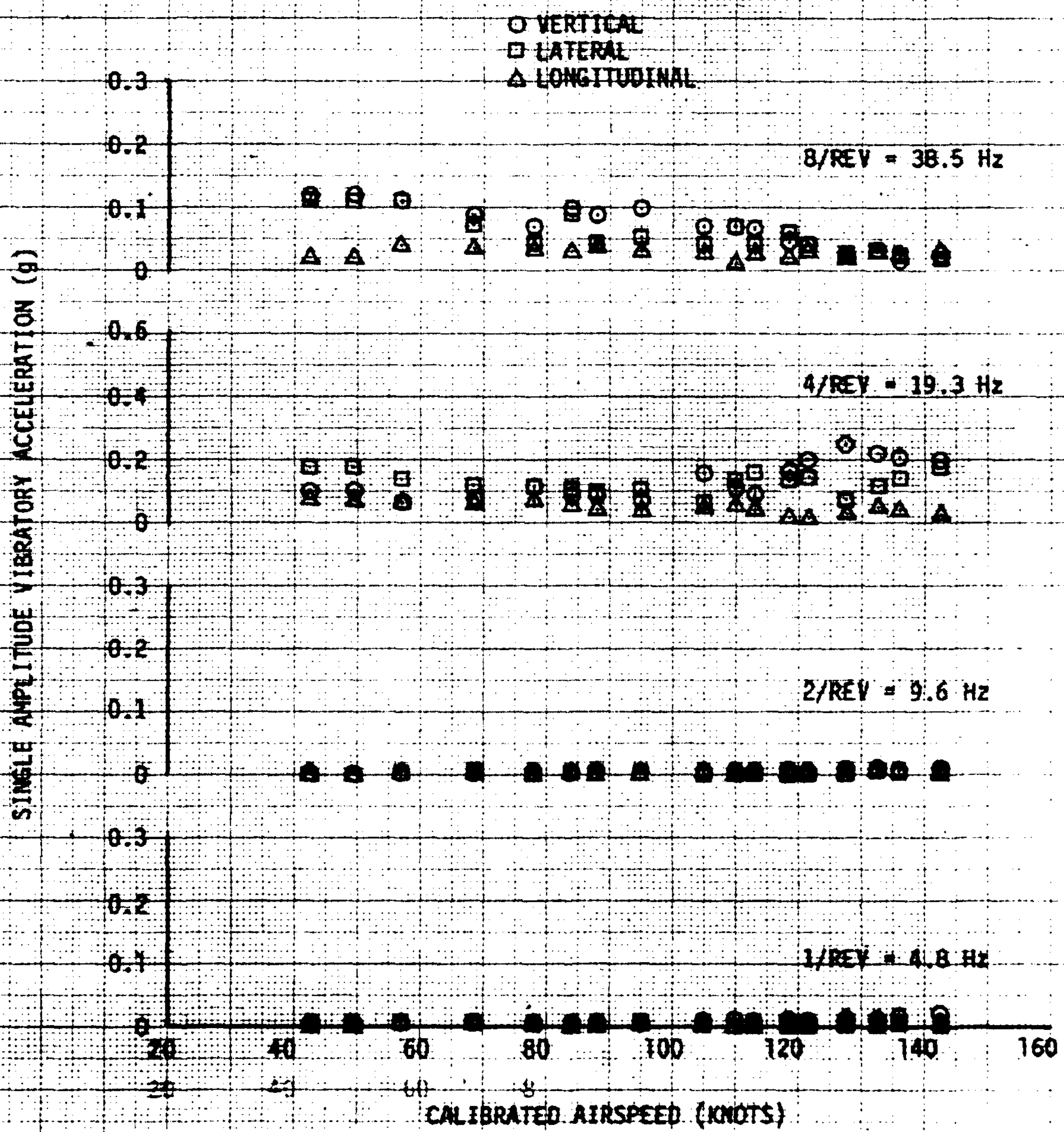


FIGURE 63
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258
COPILOT SEAT

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
14860	202.0(FWD)	0.0	5420	10.0	290	LEVEL

NOTE: 8 HELLFIRE CONFIGURATION

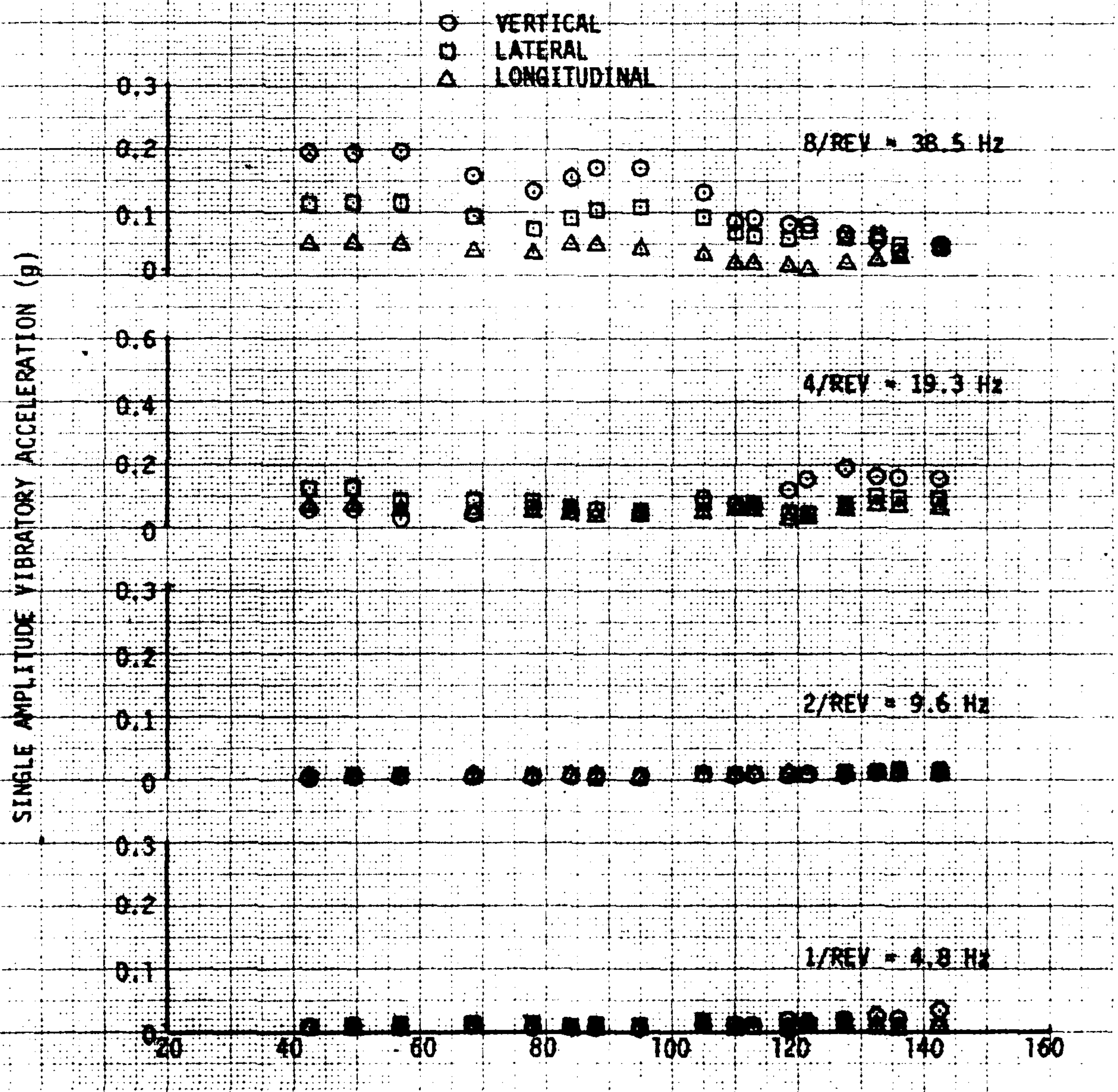


FIGURE 64
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258
PILOT SEAT

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
15740	202.0(FWD)	0.0	6380	8.0	290	LEVEL

NOTE: 8-HELLFIRE CONFIGURATION

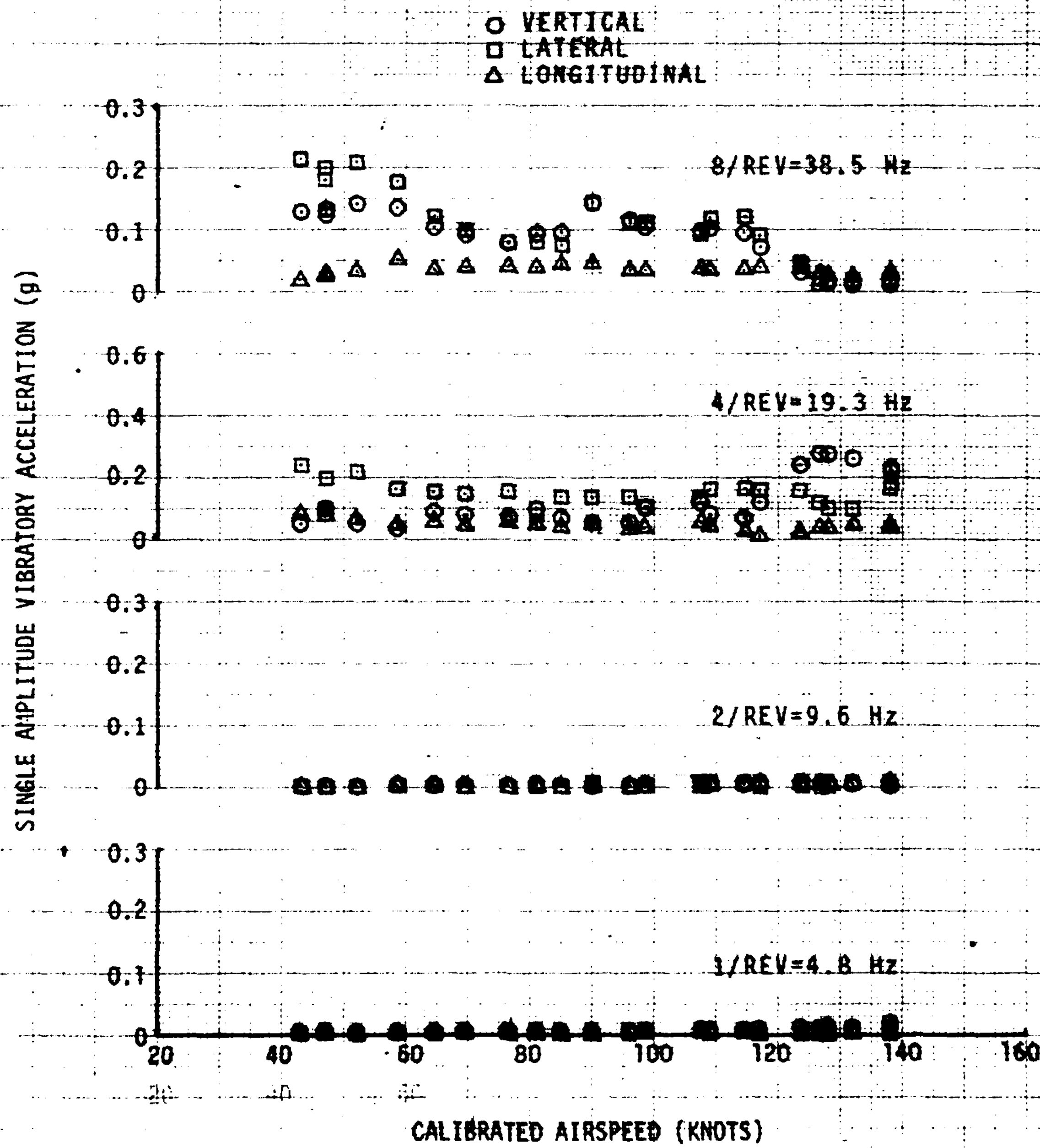


FIGURE 65
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258
COPILOT SEAT

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
15740	202.0(FWD)	0.0	6380	8.0	290	LEVEL

NOTE: 8-HELLFIRE CONFIGURATION

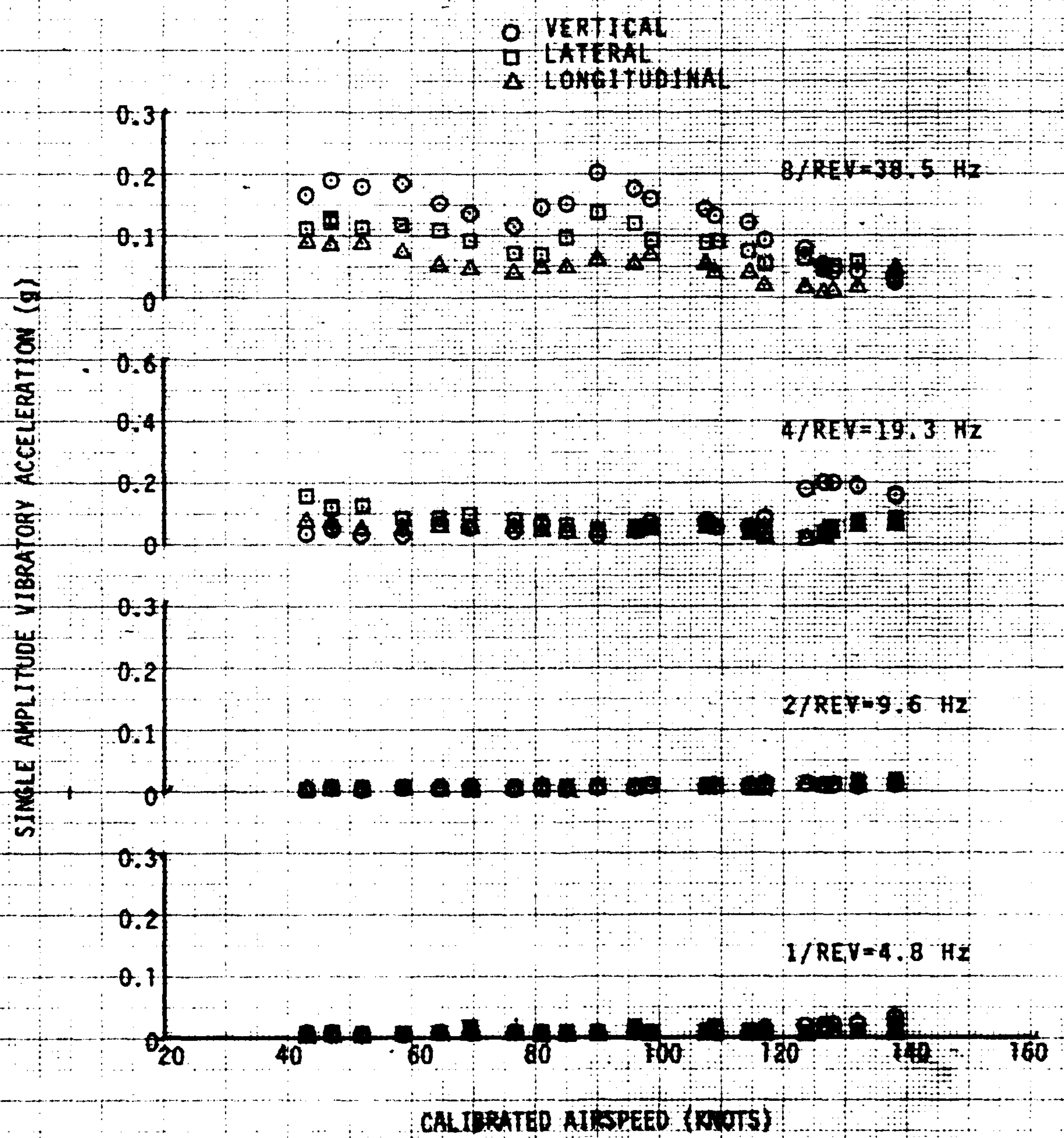


FIGURE 66
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258
PILOT SEAT

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
15640	202.2(FWD)	0.0	9720	0.5	290	LEVEL

NOTE: 8-HELLFIRE CONFIGURATION

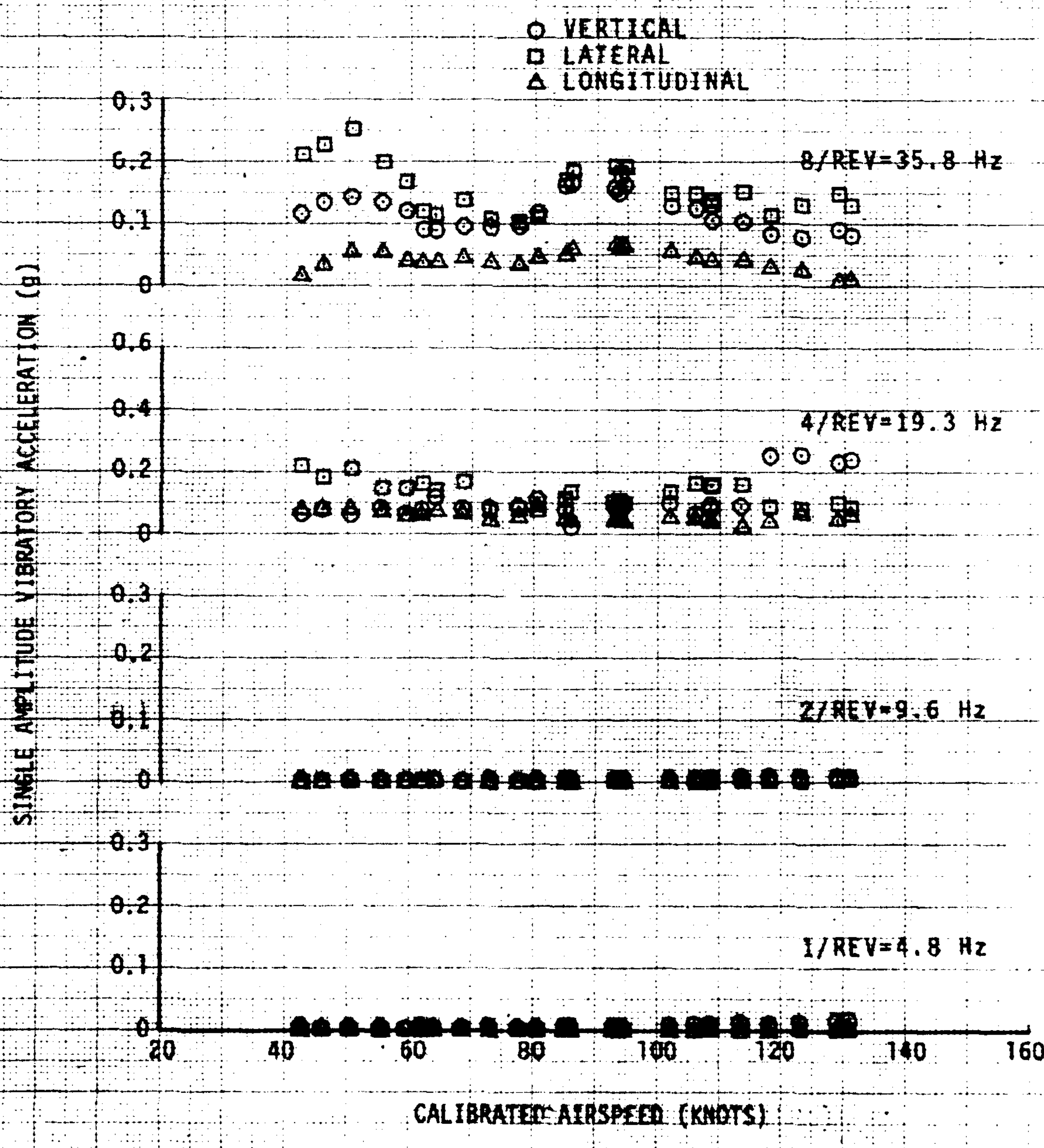


FIGURE 67
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258
COPILOT SEAT

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
15640	202.2(FWD)	0.0	9720	0.5	290	LEVEL

NOTE: 8-HELLFIRE CONFIGURATION

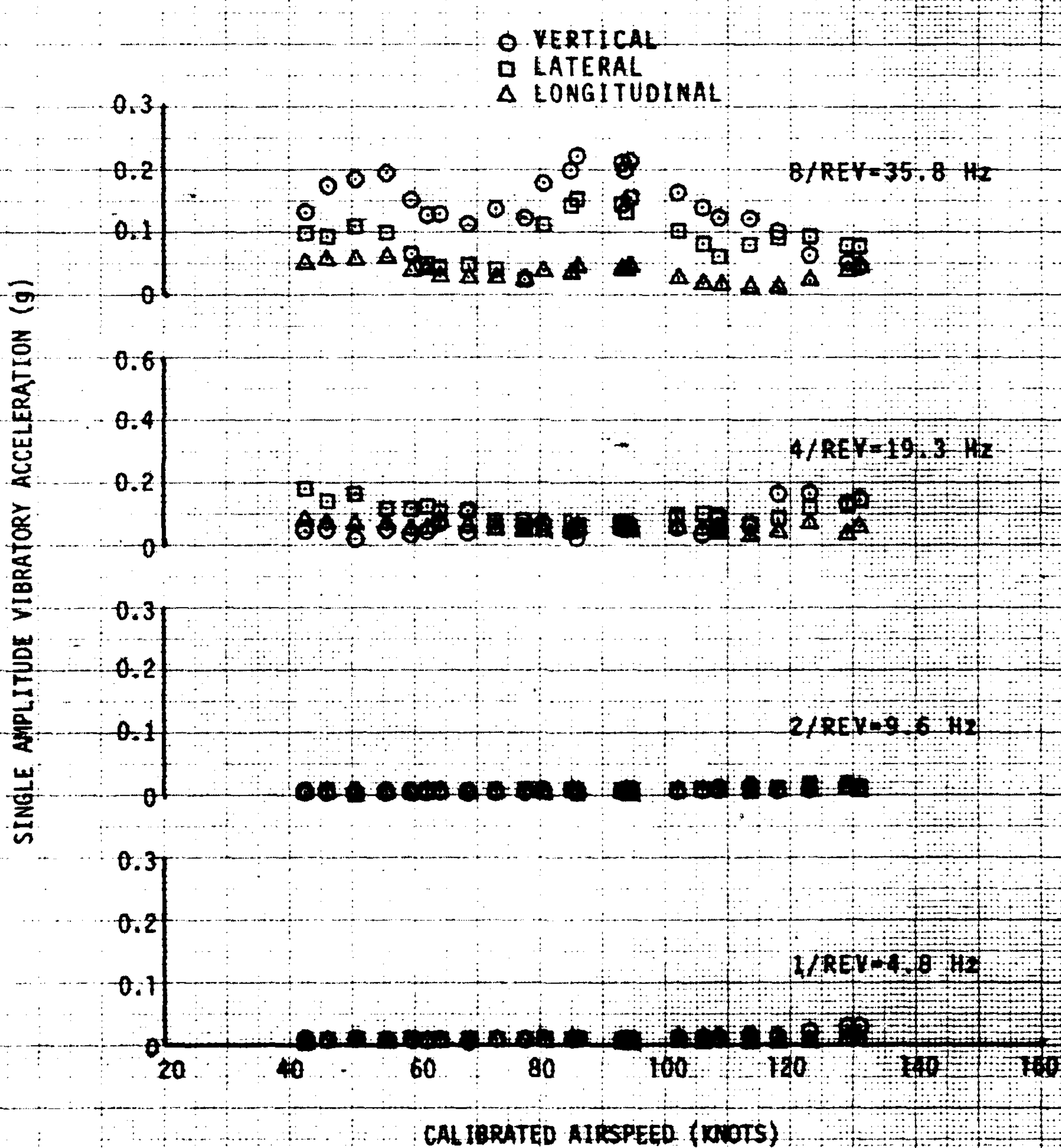


FIGURE 68
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258
PILOT FLOOR

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
15640	202.2(FWD)	0.0	9720	0.5	290	LEVEL

NOTE: 8-HELLFIRE CONFIGURATION

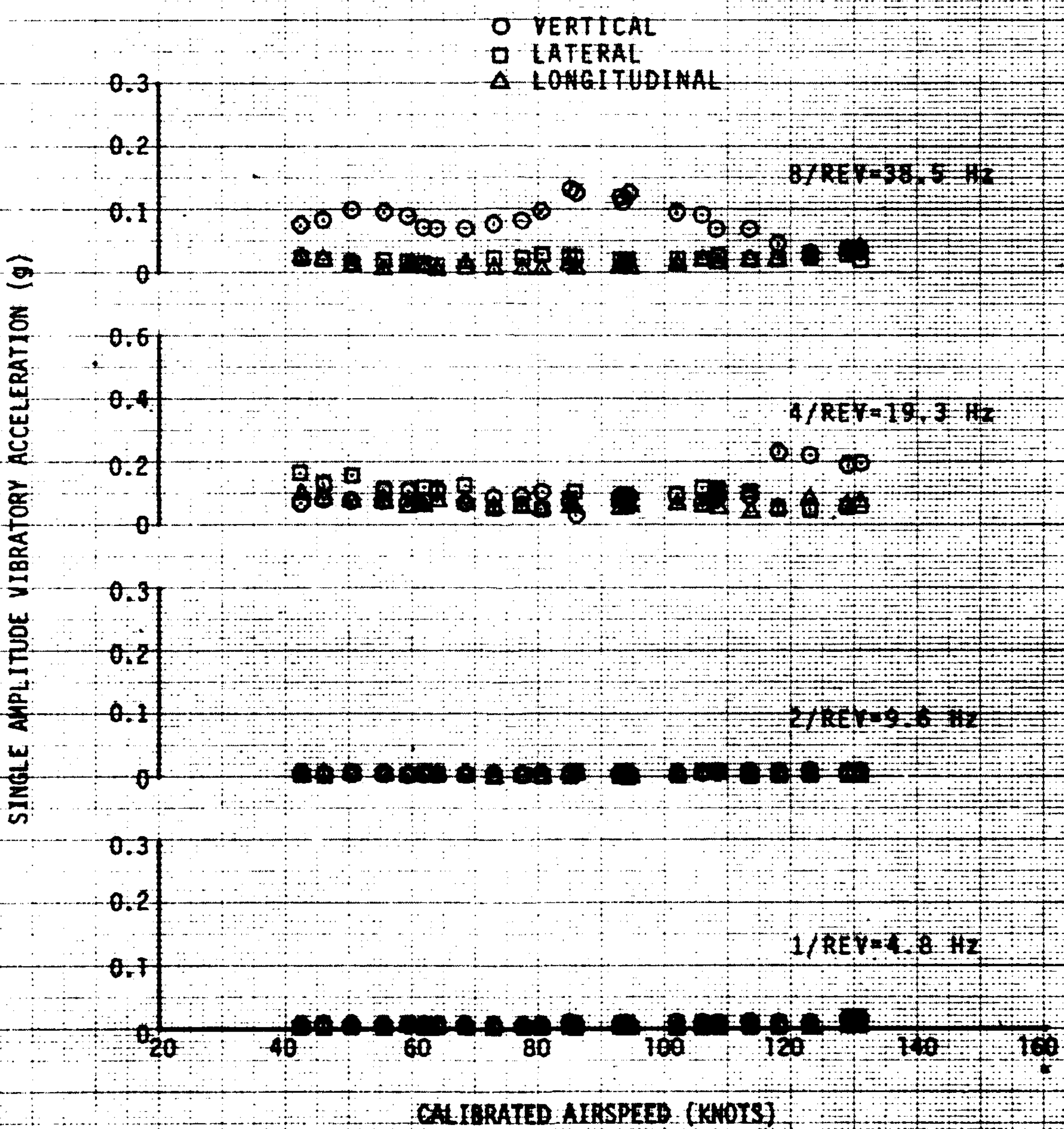


FIGURE 69
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258
COPILOT FLOOR

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
15840	202.2(FWD)	0.0	9720	0.5	290	LEVEL

NOTE: 8-HELLFIRE CONFIGURATION

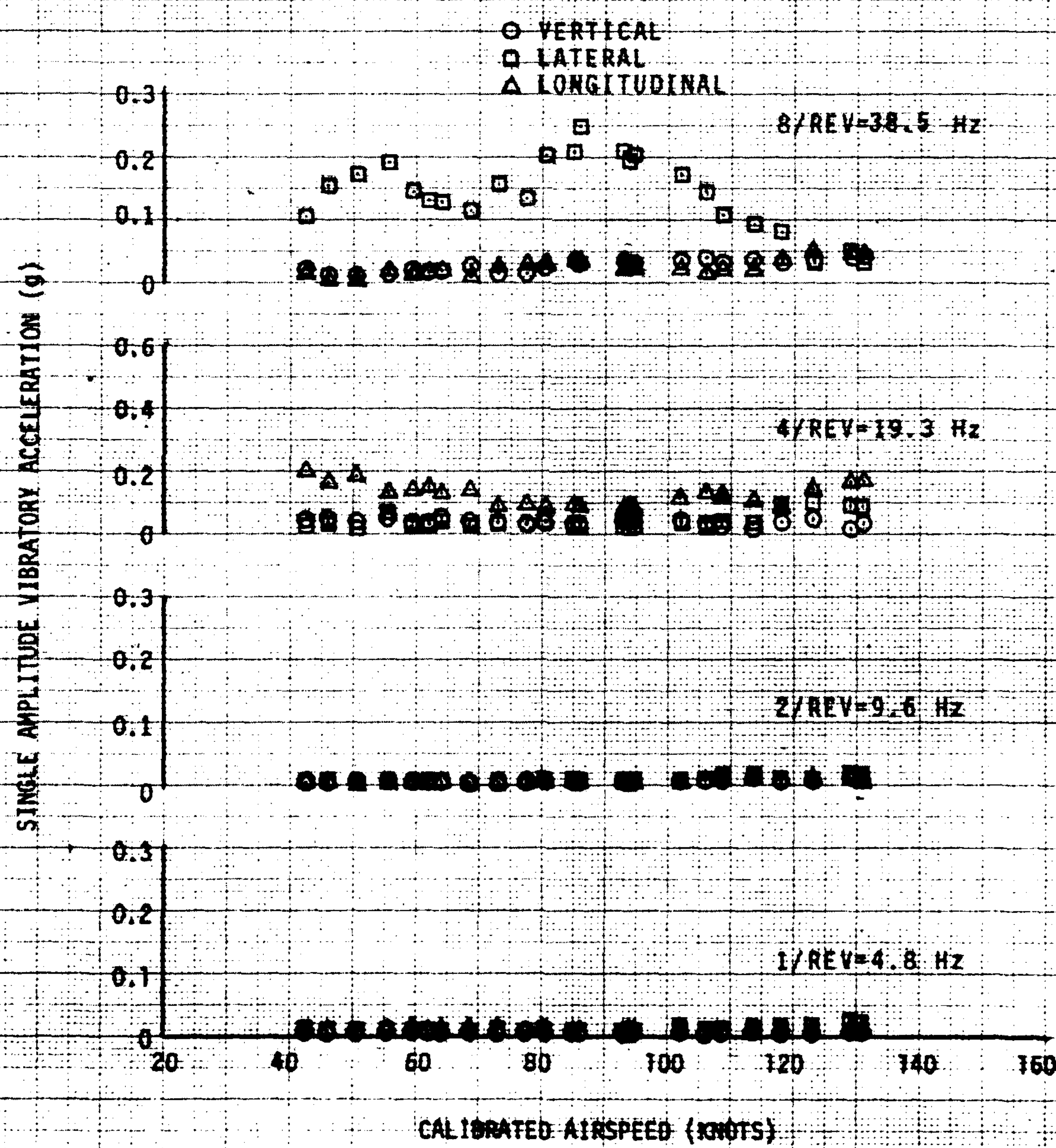


FIGURE 70
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258
AIRCRAFT CG

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
15640	202.2(FWD)	0.0	9720	0.5	290	LEVEL

NOTE: 8-HELLFIRE CONFIGURATION

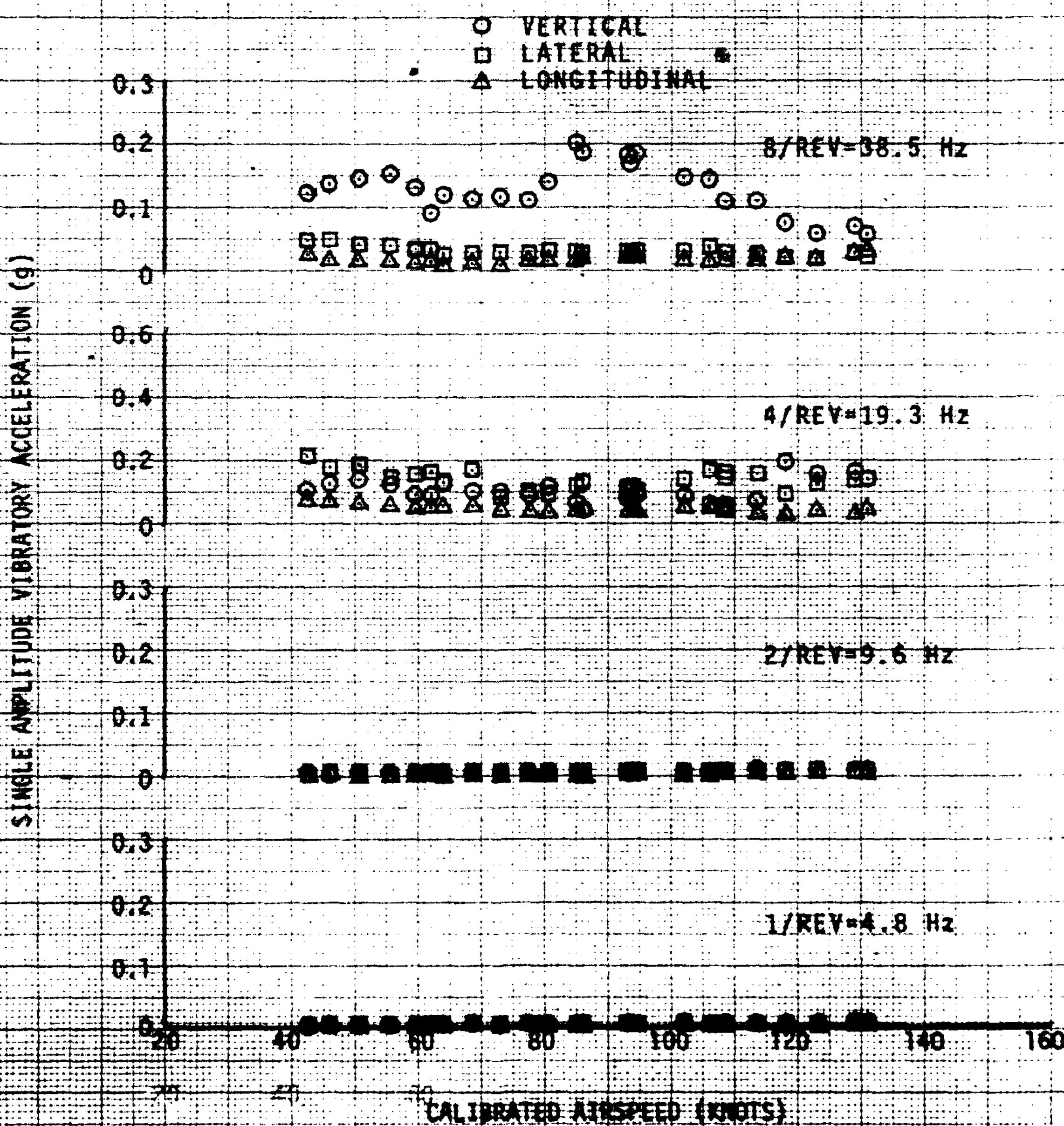


FIGURE 71
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258

PILOT SEAT

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
15740	205.2(AFT)	0.0	5020	6.5	290	LEVEL FLIGHT

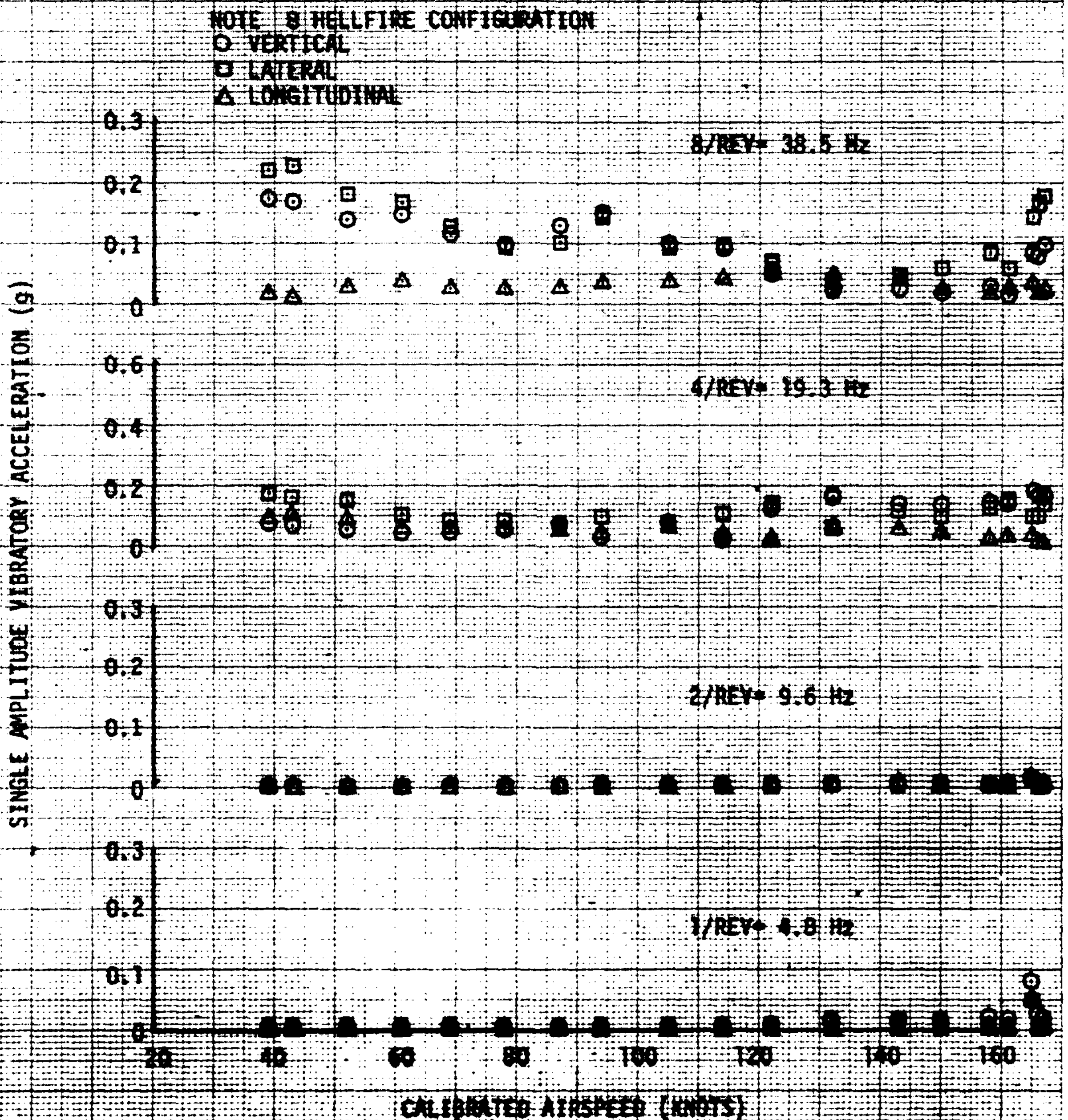


FIGURE 72
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258
CO PILOT SEAT

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
15740	205.2(AFT)	0.0	5020	6.5	290	LEVEL FLIGHT

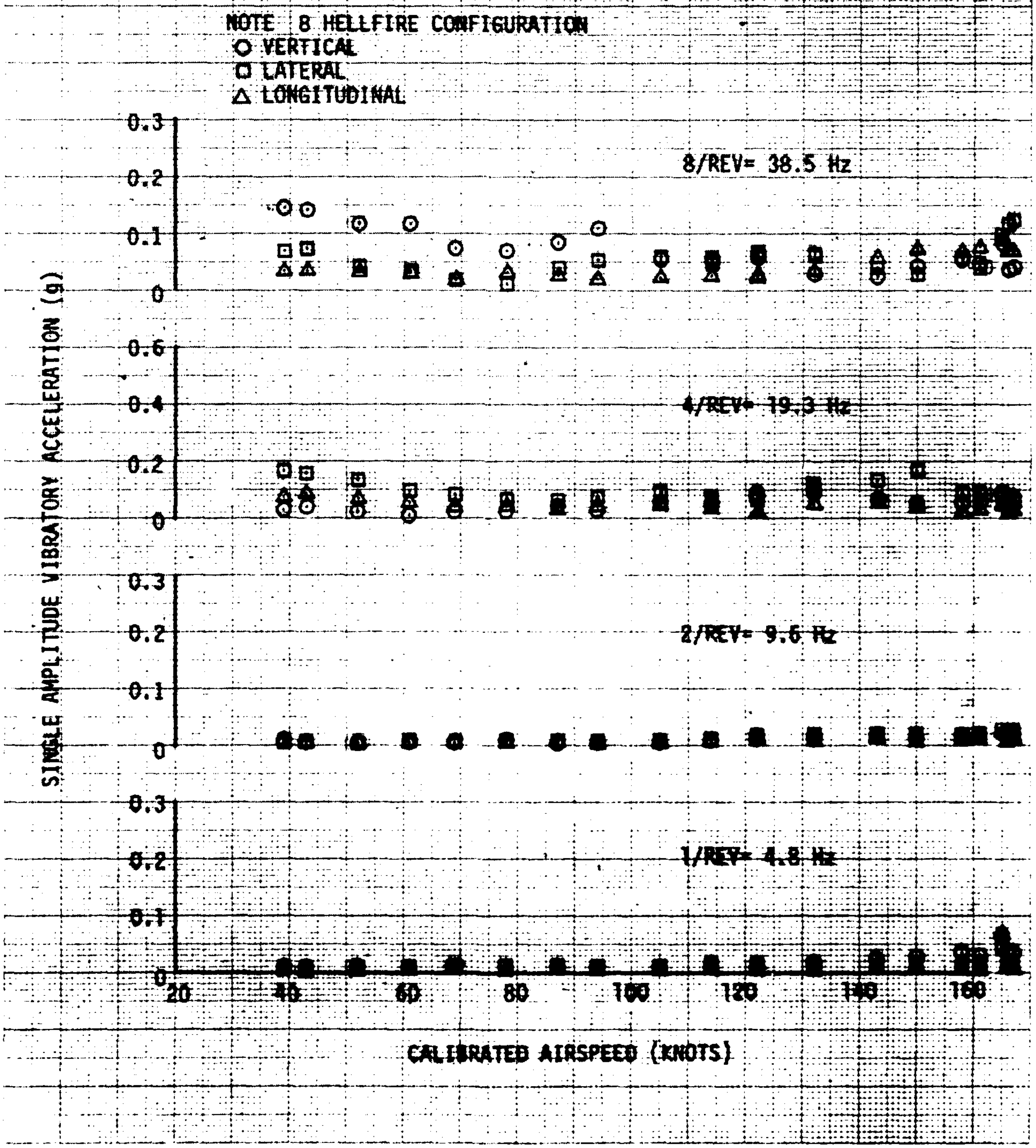


FIGURE 73
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258
PILOT SEAT

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
14400	205.7(AFT)	0.0	5840	8.5	290	LEVEL

NOTES: 1. 8-HELLFIRE CONFIGURATION
2. VERTICAL VIBRATION ABSORBER REMOVED

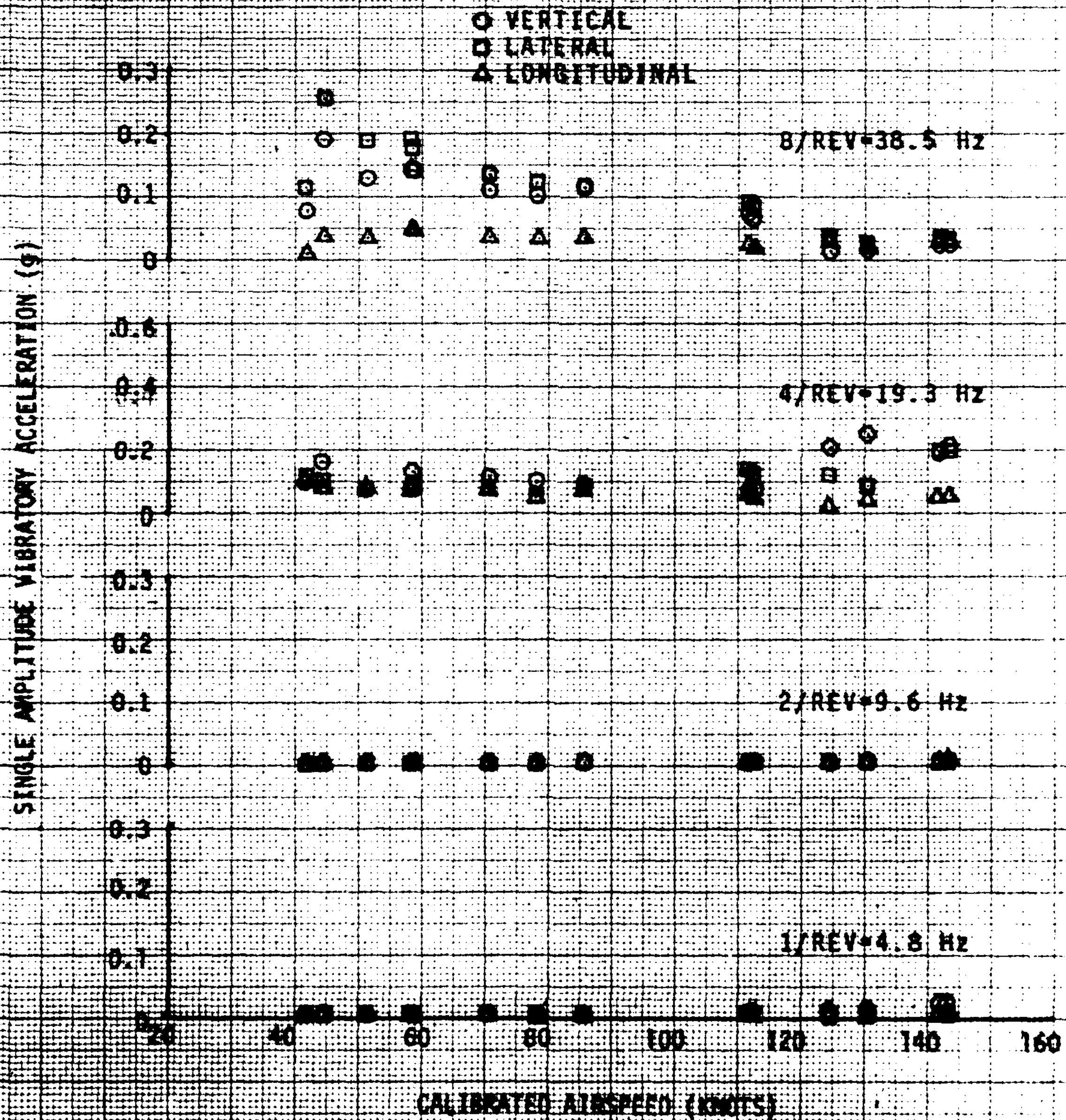


FIGURE 76
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258
COPILOT FLOOR

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
14400	205.7(AFT)	0.0	5840	8.5	290	LEVEL FLIGHT

NOTES: 1. 8 HELLFIRE CONFIGURATION
2. VERTICAL VIBRATION ABSORBER REMOVED

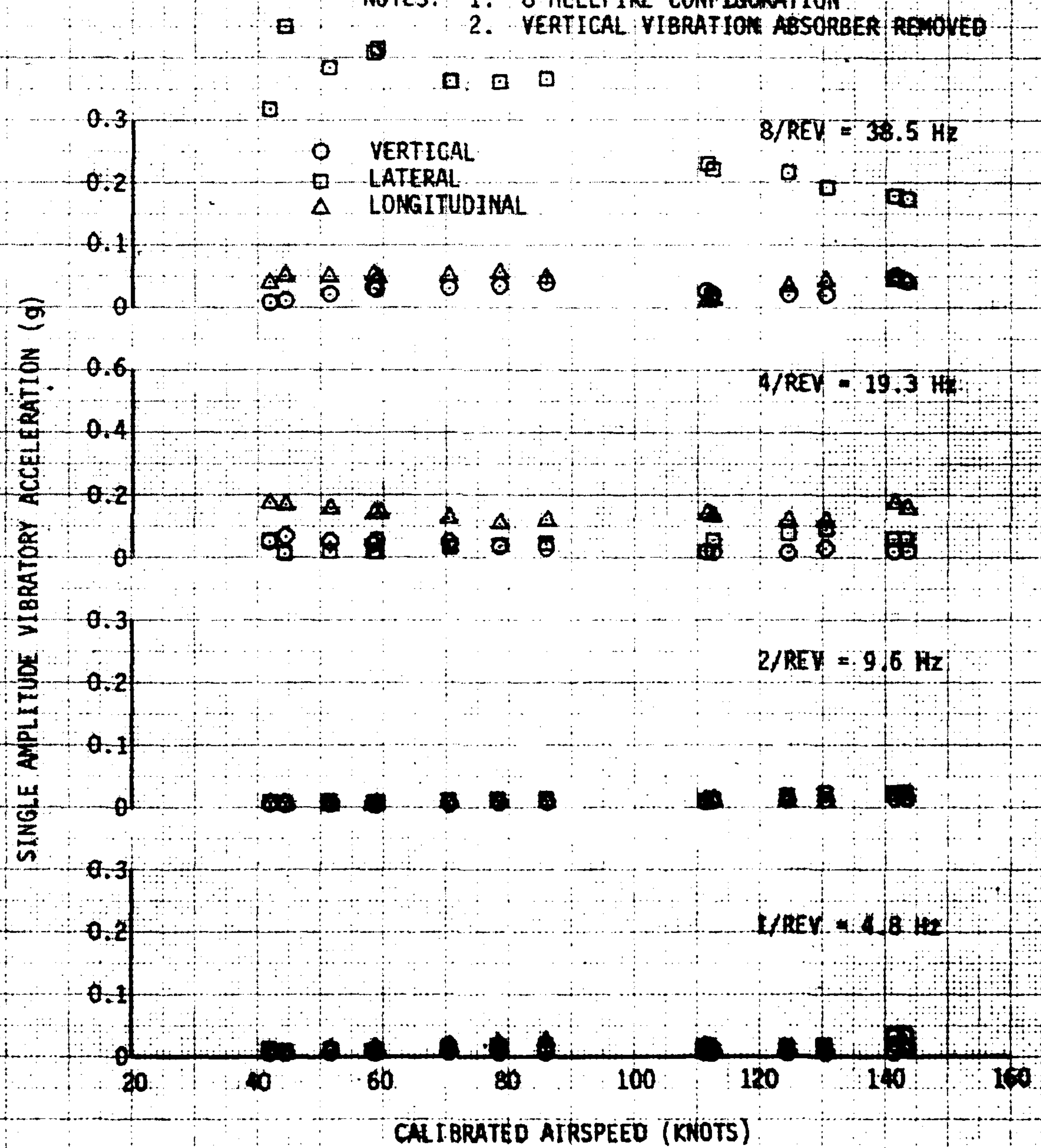


FIGURE 77
VIBRATION CHARACTERISTICS
YAH-64 USA S/N [illegible]
AIRCRAFT CG

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
14400	205.7(AFT)	0.0	5840	8.5	290	LEVEL FLIGHT

NOTES: 1. 8 HELLFIRE CONFIGURATION
2. VERTICAL VIBRATION ABSORBER REMOVED

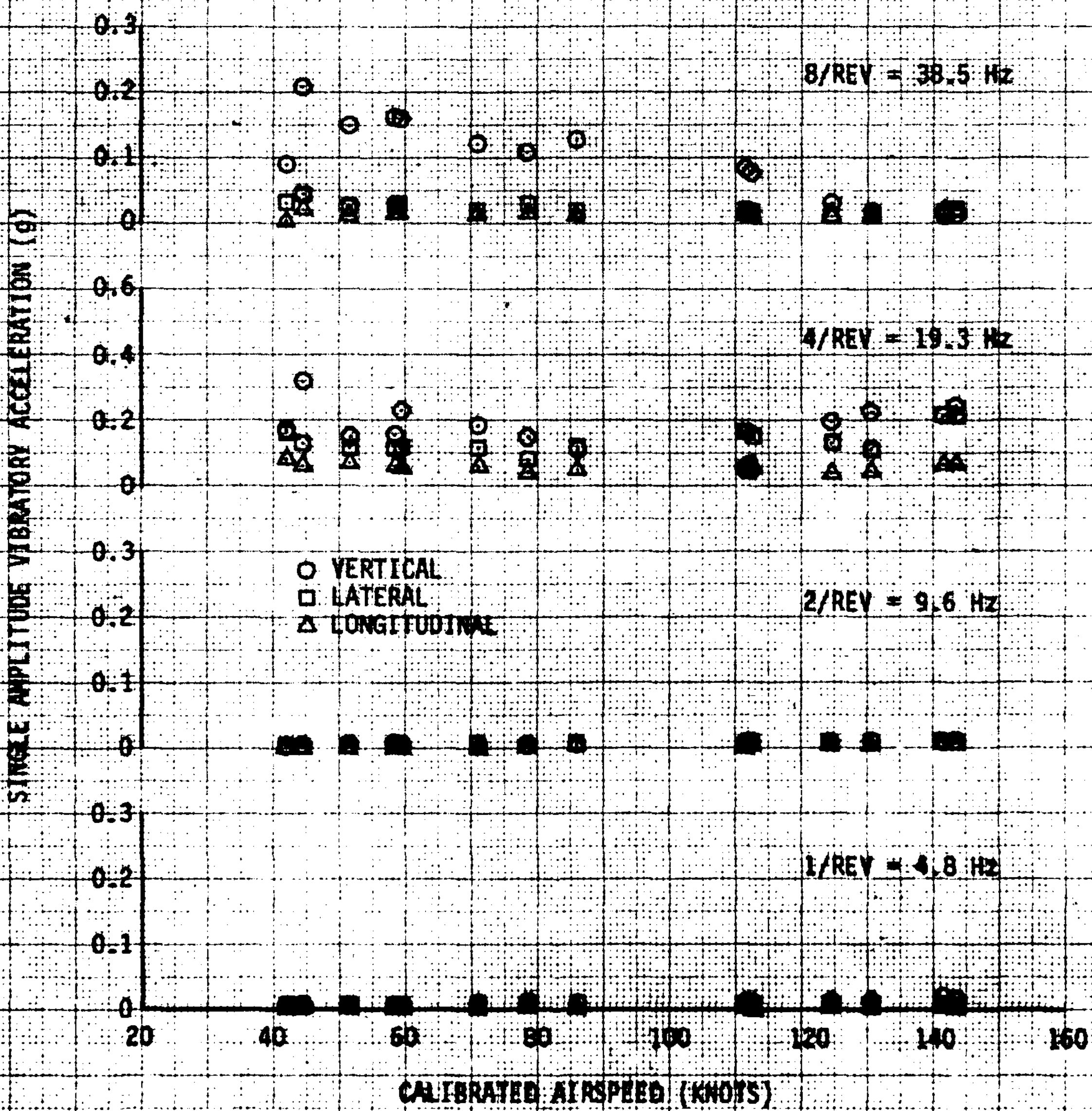


FIGURE 74
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258
COPILOT SEAT

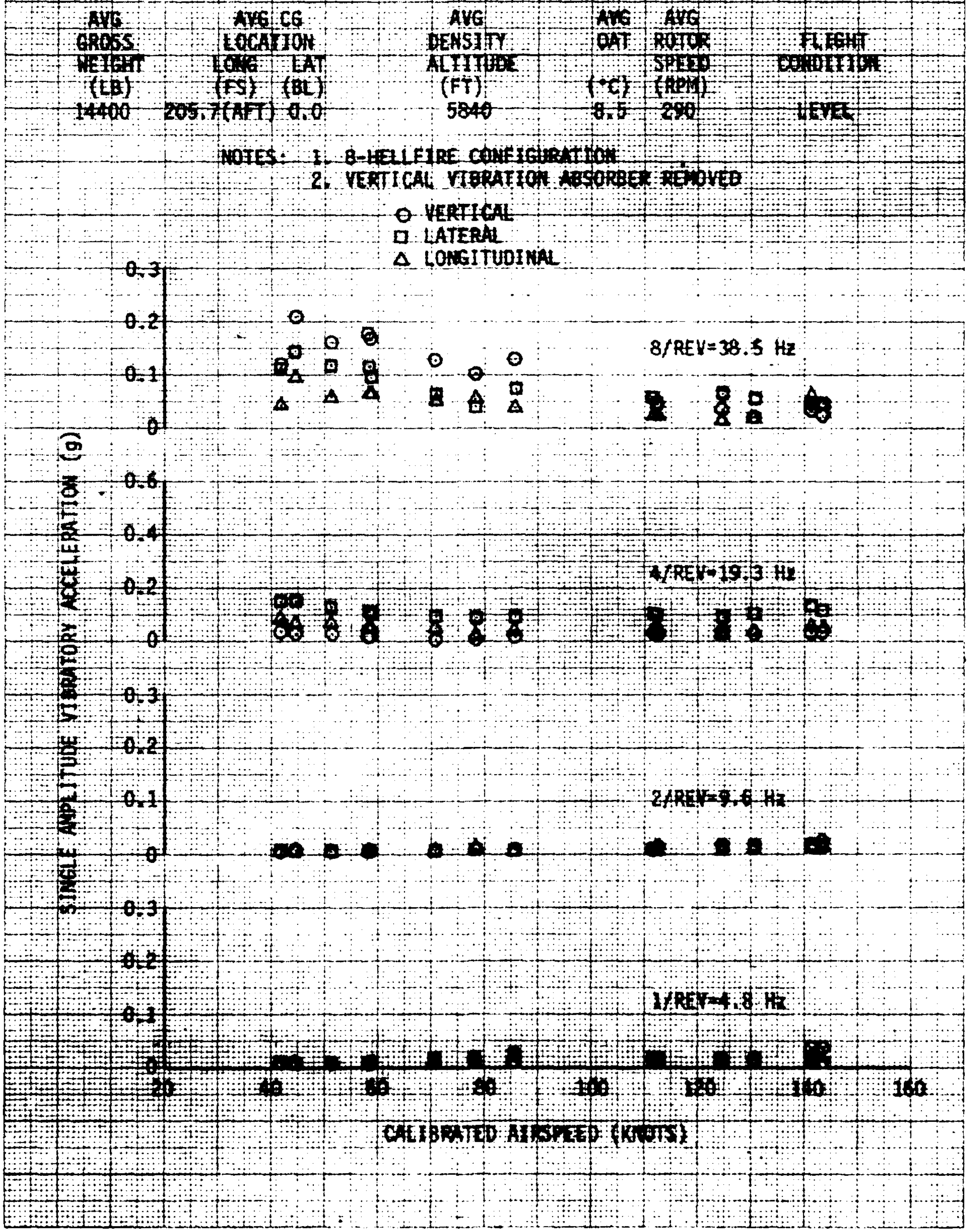
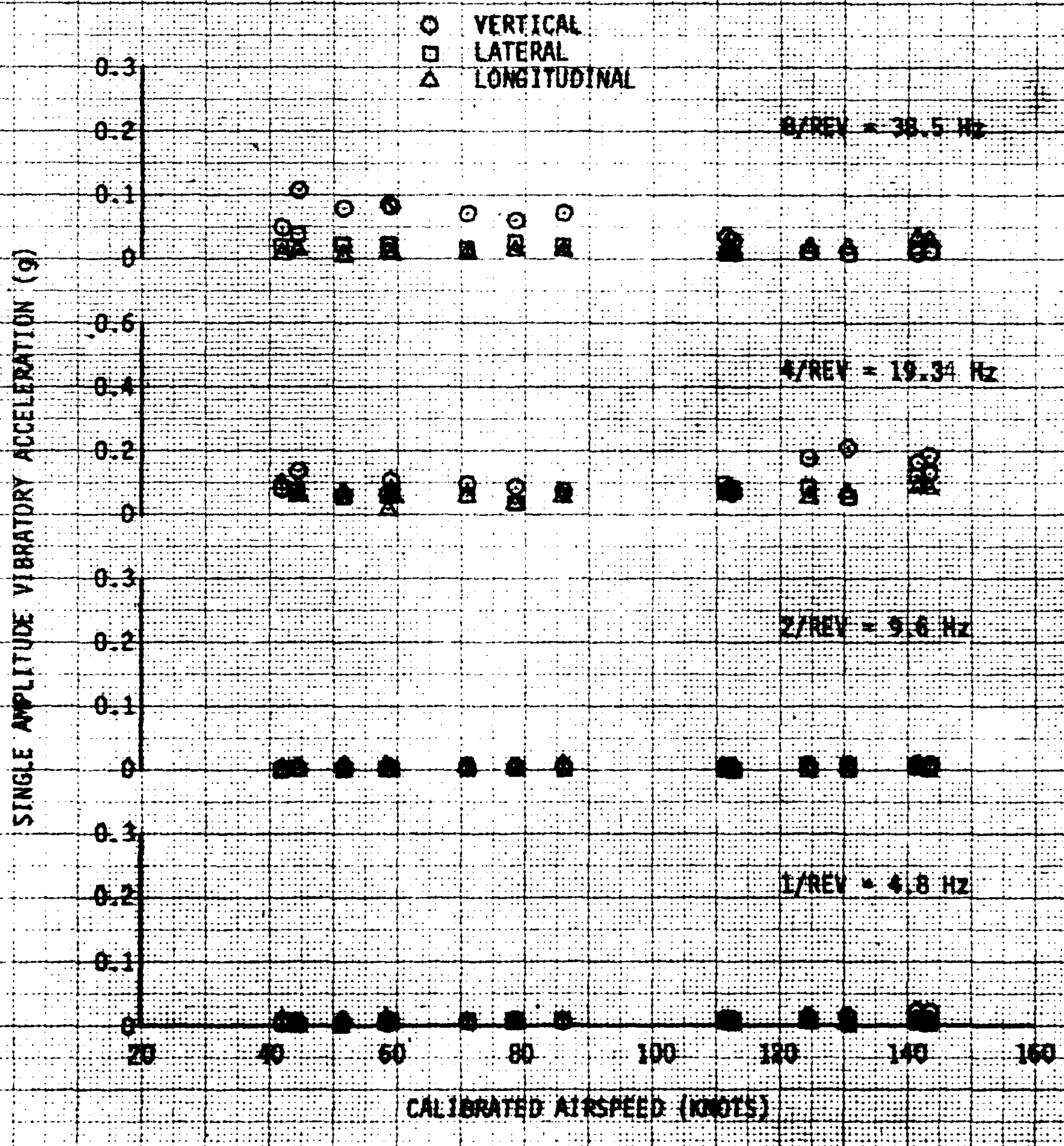


FIGURE 75
VIBRATION CHARACTERISTICS
YAH-64 USA S/N 77-23258
PILOT FLOOR

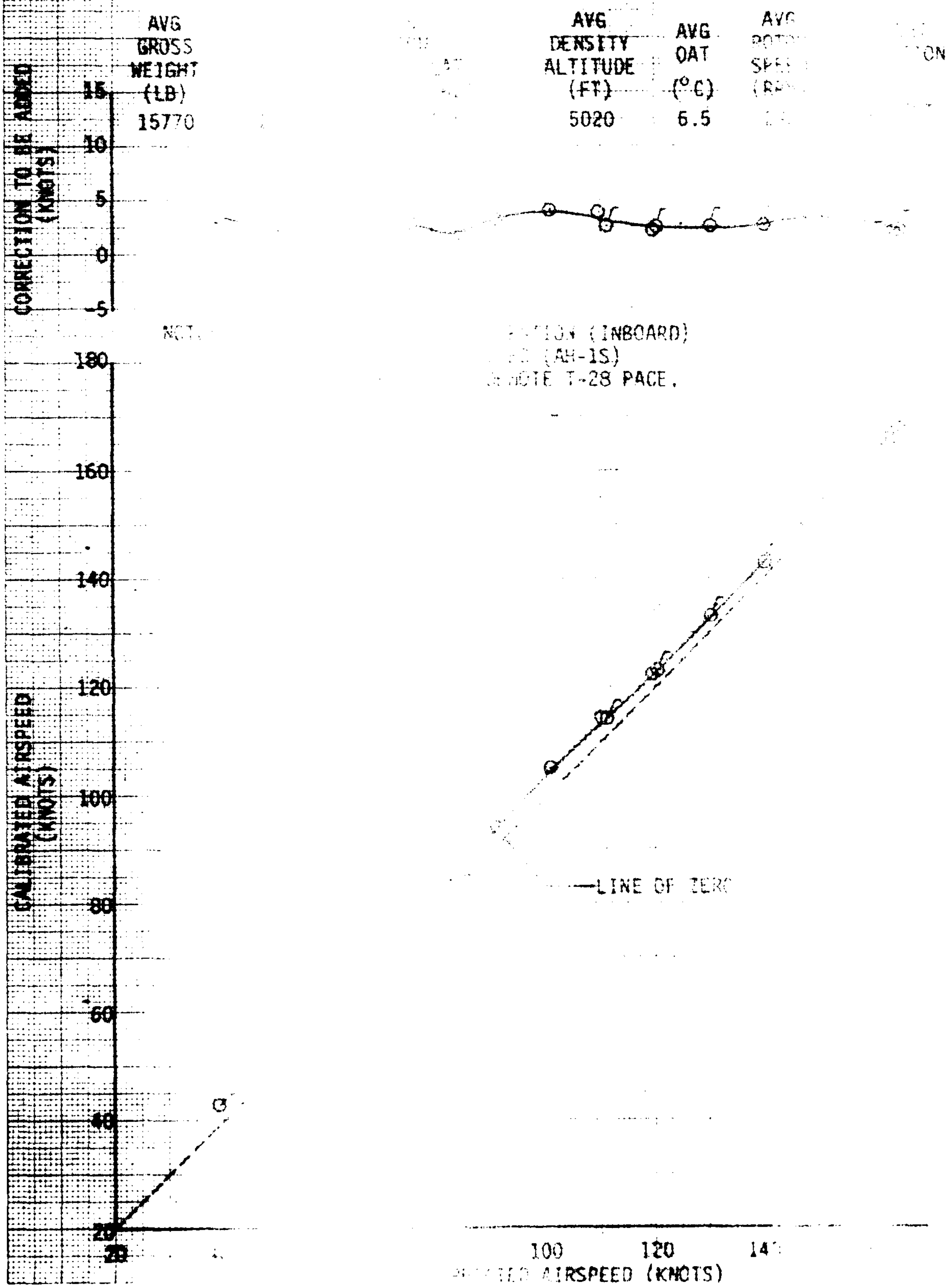
AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
14400	205.7(AFT)	0.0	5840	8.5	290	LEVEL FLIGHT

NOTES: 1. 8 HELLFIRE CONFIGURATION
2. VERTICAL VIBRATION ABSORBER REMOVED



RATION RIGHT WING

77-23258



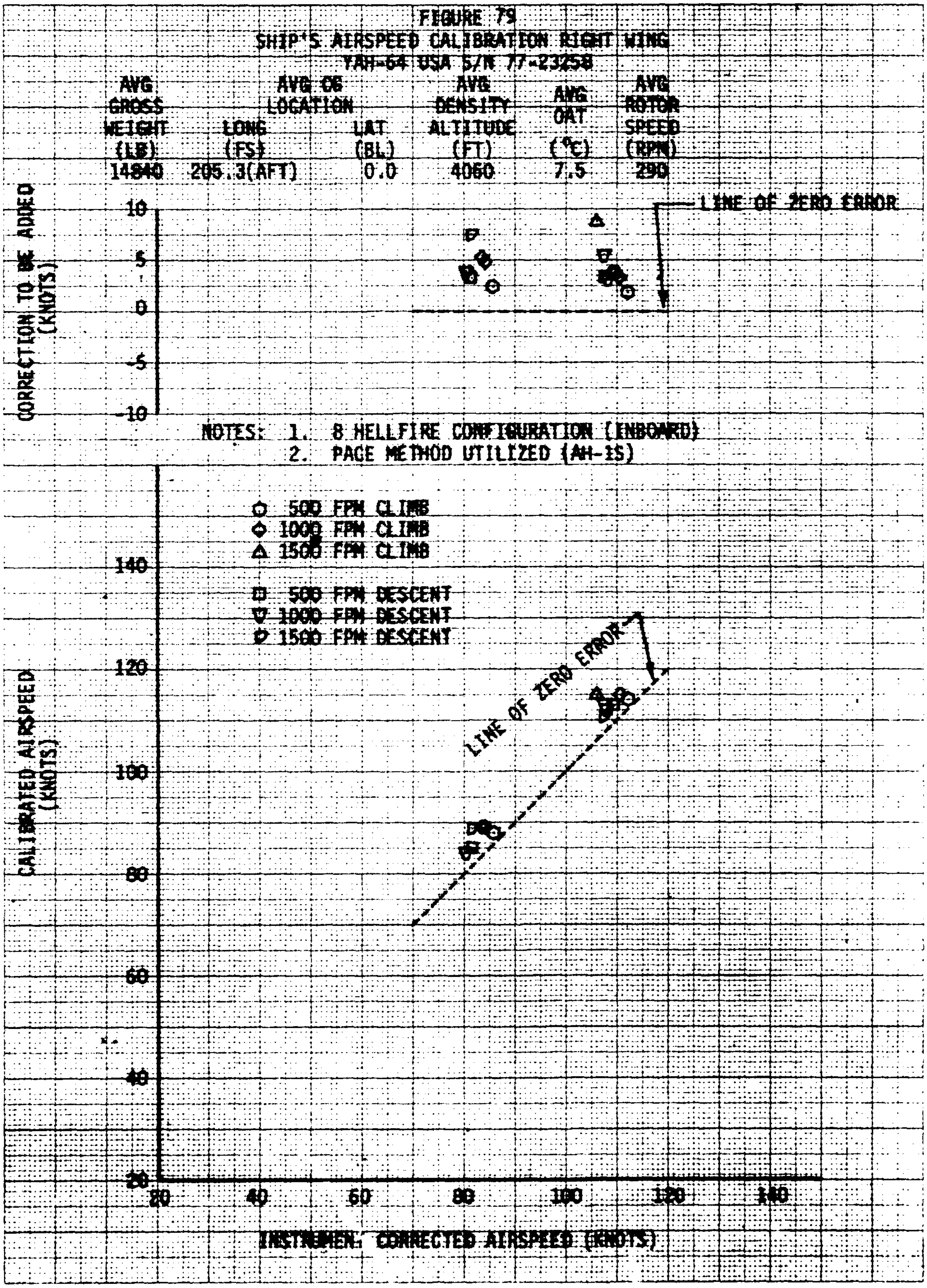


FIGURE 79
SHIP'S AIRSPEED CALIBRATION RIGHT WING
YAH-64 USA S/N 77-23258
AVG GROSS WEIGHT (LB) 14840
AVG CG LOCATION LONG (FS) 205.3(AFT)
LAT (BL) 0.0
AVG DENSITY ALTITUDE (FT) 4060
AVG OAT (°C) 7.5
AVG ROTOR SPEED (RPM) 290
CORRECTION TO BE ADDED (KNOTS)
LINE OF ZERO ERROR
NOTES: 1. 8 HELLFIRE CONFIGURATION (INBOARD)
2. PACE METHOD UTILIZED (AH-1S)
500 FPM CLIMB
1000 FPM CLIMB
1500 FPM CLIMB
500 FPM DESCENT
1000 FPM DESCENT
1500 FPM DESCENT
LINE OF ZERO ERROR
CALIBRATED AIRSPEED (KNOTS)
INSTRUMENT CORRECTED AIRSPEED (KNOTS)

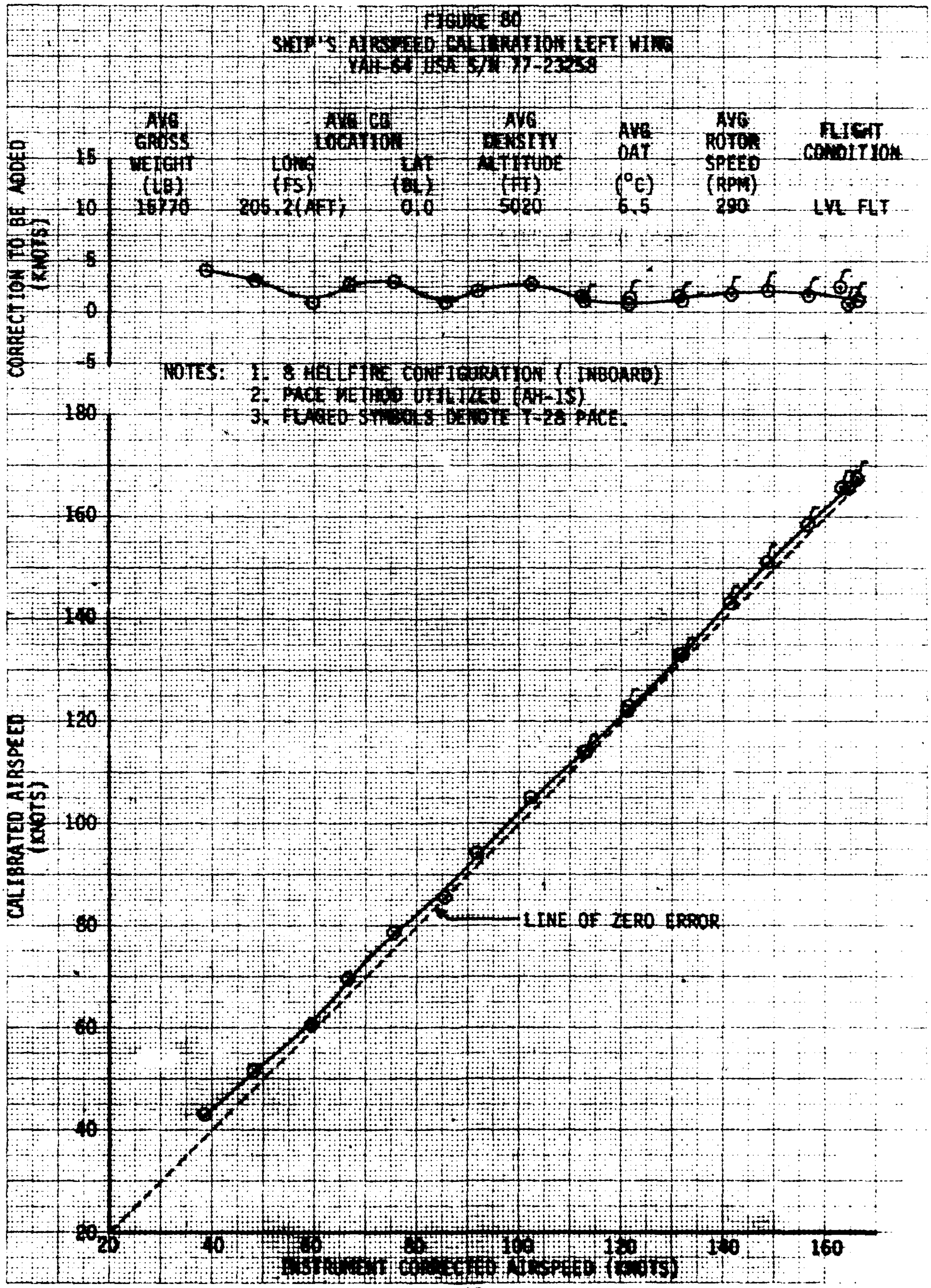


FIGURE 80
SHIP'S AIRSPEED CALIBRATION LEFT WING
YAH-64 USA S/N 77-23258
AVG GROSS WEIGHT (LB) 15770
AVG CG LOCATION LONG (FS) 205.2(AFT) LAT (BL) 0.0
AVG DENSITY ALTITUDE (FT) 5020
AVG OAT (°C) 6.5
AVG ROTOR SPEED (RPM) 290
FLIGHT CONDITION LVL FLT
CORRECTION TO BE ADDED (KNOTS)
NOTES: 1. 8 HELLFIRE CONFIGURATION (INBOARD)
2. PACE METHOD UTILIZED (AH-1S)
3. FLAGED SYMBOLS DENOTE T-28 PACE.
LINE OF ZERO ERROR
CALIBRATED AIRSPEED (KNOTS)
INSTRUMENT CORRECTED AIRSPEED (KNOTS)

APPENDIX F. GLOSSARY

a	Speed of sound
A	Main rotor disc area (ft^2)
AAH	Advance Attack Helicopter
ADS	Air Data System
app	Appendix
APU	Auxiliary Power Unit
AVRADCOM	US Army Aviation Research and Development Command
BL	Butt Line
BUCS	Back-up Control System
C	Celsius
CAS	Control Augmentation System
cg	Center of Gravity
C_L	Centerline
C_P	Coefficient of Power
CPG	Copilot/gunner
C_T	Coefficient of Thrust
DASE	Digital Automatic Stabilization Equipment
DASEC	Digital Automatic Stabilization Equipment Computer
deg	Degree
DT-1	Development Test 1
EADI	Electronic Attitude Direction Indicator
ECRR	Engineering Change Request and Record
ECU	Electrical Control Unit
EDT-1	Engineer Design Test 1
EDT-2	Engineer Design Test 2
EMI	Electromagnetic Interference
ENCU	Environmental Control Unit
EPR	Equipment Performance Report
ETP	Experimental Test Procedure
FABS	Forward Avionics Bays
fig	Figure
FOD	Foreign Object Damage
fs, FS	Fuselage Station
ft	Feet
HARS	Heading and Attitude Reference System
HAS	Hover Augmentation System
GCT	Government Competitive Test
GPM	Gallons Per Minute
GW	Gross Weight
HH	Hughes Helicopters
HMU	Hydromechanical Unit
H_P	Pressure altitude
HQRS	Handling Qualities Rating Scale
Hz	Hertz
IGE	In Ground Effect
IMC	Instrument Meteorological Conditions
in.	Inches
IR	Infrared
IRP	Intermediate Rated Power
KCAS	Knots Calibrated Airspeed
KIAS	Knots Indicated Airspeed
KTAS	Knots True Airspeed
lb	Pound

LVDT	Linear Variable Displacement Transducer
M_{tip}	Advancing tip mach number
NAMPP	Nautical air miles per pound of fuel
N_G	Gas producer speed
NOE	Nap of the Earth
N_P	Power turbine speed
N_R	Main rotor speed
OAT	Outside air temperature
OGE	Out of Ground Effect
OT-II	Operational Test II
PCM	Pulse Code Modulation
PIO	Pilot Induced Oscillation
PNVS	Pilot Night Vision System
psi	Pounds per square inch
psig	Pounds per square inch gauge
Q	Engine output shaft torque
R	Radius (ft)
ref	Reference
rpm	Revolutions per minute
SAS	Stability Augmentation System
SCAS	Stability and Control Augmentation System
SCU	Stabilator Control Unit
sec	Seconds
SHP, shp	Shaft Horsepower
S/N	Serial Number
TADS	Target Acquisition and Designation System
TFS	Trim Feel System
TGT	Turbine gas temperature
$T^{4.5}$	Turbine gas temperature
USAAEFA	US Army Aviation Engineering Flight Activity
V	Velocity
VDC	Volts Direct Current
V_H	Maximum Horizontal Velocity
VMC	Visual Meteorological Conditions
VNE	Never exceed airspeed
V_T	True airspeed
VRS	Vibration Rating Scale
WL	Water line
wf	Fuel flow rate

Greek and Miscellaneous Symbols

Δ	Incremental change
μ	Advance ratio
ρ	Air density ($slugs/ft^3$)
σ	Air density ratio
Ω	Main rotor angular velocity (radians/sec)
~	Approximately
4/rev	4th harmonic of the main rotor

APPENDIX G. EQUIPMENT PERFORMANCE REPORTS

The following EPRs were submitted:

NUMBER	SUBJECT
80-03-1	Fuel transfer pump
80-03-2	TADS/PNVS mounting bracket
80-03-3	Rotor brake system
80-03-4	Collective friction assembly
80-03-5	External power monitor
80-03-6	Stabilator control panel
80-03-7	Engine nose gearbox leak
80-03-8	Master caution light
80-03-9	Engine/rotor speed indicator
80-03-10	Heading attitude reference system
80-03-11	Stabilator control unit
80-03-12	Hydraulic fluid sample procedure
80-03-13	Main rotor lead/lag bearings
80-03-14	Utility hydraulic manifold bleed/overservice
80-03-15	Main transmission leak
80-03-16	Impending bypass button, nose gearbox
80-03-17	Main transmission input clutch assembly seal leak
80-03-18	Stabilator bearings (bushings)
80-03-19	Main transmission impending bypass buttons (2 each)

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US Army Aviation Research and Development Command (DRDAV-DI, DRDAV-EE, DRDAV-EG)	10
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Advanced Attack Helicopter (DRCPM-AAH-TM, DRCPM-AAH-APM-TE)	6
General Electric - AEG (Mr. Koon)	1
Hughes Helicopters (Mr. Gerry Ryan)	3
US Army Missile Command (DRCPM-HDT-T)	1